**Small Satellite Research Center**
**United States Air Force Academy**

**USAFA-FS2-FSDP-23**
**Revision 5**
**Nov 2002**

# FLIGHT SAFETY DATA PACKAGE

## PHASE 2/3

FALCONSAT-2

Small Satellite Research Center
Department of Astronautics
U.S. Air Force Academy, CO 80840

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# SIGNATURES

**Customer Payload:** FS-2 Spacecraft

**Customer:** U.S. Air Force Academy (USAFA), Space Test Program (STP)

**Date:** November 2002

**Customer Approval:**

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**REVISIONS**

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# 1. INTRODUCTION

## 1.1 Purpose

This document comprises the Phase 2/3 Flight Safety Data Package (FSDP) for the FalconSat-2 (FS-2) satellite being developed by the United States Air Force Academy (USAFA). FS-2 is manufactured to fly aboard the shuttle for deployment as part of the Hitchhiker program. FS-2 will be installed into a Hitchhiker (HH) canister using the Pallet Ejection System (PES). The canister will have a Hitchhiker Motorized Door Assembly (HMDA) opening lid. FS-2, with its Miniature Electrostatic Analyzer (MESA) experiment, is a nano-satellite designed to investigate ionospheric plasma depletions that cause radio transmission disruptions by conducting *in situ* sampling of plasma density.

FS-2 is designed to be inactive while installed in the Shuttle. This is ensured by the use of four series-connected micro-switches that inhibit power from the solar panels to the spacecraft batteries while in the launch configuration. Once deployed from the Shuttle, FS-2 will begin operation by initializing its hardware, charging its batteries, and preparing for contact with USAFA's ground station after a minimum of three orbits.

The purpose of this FSDP is to provide the Shuttle program a comprehensive safety assessment of FS-2 subsystems that potentially pose hazards to the Shuttle. This FSDP is intended to enumerate all potential hazards associated with the FS-2 payload and provide sufficient information to show that FS-2 complies with the requirements specified in NSTS 1700.7B.

## 1.2 Scope

This FSDP encompasses all subsystems associated with the FS-2 spacecraft as a Shuttle secondary payload. FS-2 will use Hitchhiker government furnished equipment (GFE). NASA GFE includes the PES adapter ring, Hitchhiker canister, canister heaters and base plate thermistor, avionics package, and Pallet Ejection System. FS-2 will be mounted inside the Hitchhiker canister via the PES. NASA/GSFC will provide the mating launch vehicle interface (LVI) with the prescribed dimensional configuration as defined in the *Hitchhiker Customer Accommodations & Requirements Specification,* NASA/GSFC 740-SPEC-008, 1999, pp. 1-250 thru 2-148.

The FS-2 payload is divided into five subsystems plus ground support equipment (GSE). The FS-2 subsystems include:

- Structure
- Electrical Power Subsystem (EPS)
- Communication Subsystem (COM)
- Data Handling Subsystem
- Miniature Electrostatic Analyzer (MESA) experiment

The FS-2 GSE will be used at all the Hitchhiker integration sites, both at NASA/GSFC and at NASA/KSC and may be required at non-Hitchhiker sites such as the OPF or pad (battery charging only). The FS-2 GSE will be used to accomplish functional and verification testing; charge/discharge and recondition the flight battery; and to verify status of safety inhibits. Specific operations are discussed later in this FSDP.

## 1.3 Customer Data

FS-2 customer data are listed in **Error! Reference source not found.**.

**Table 1: Customer Data**

| | |
|---|---|
| Customer Payload Name | FalconSAT-2 Spacecraft |
| Customer Payload Acronym | FS-2 |
| Customer Information | Small Satellite Research Center<br>Department of Astronautics<br>US Air Force Academy, CO, 80840 |
| Program Manager<br>(USAFA) | LtCol Jerry Sellers<br>719-333-3315, Jerry.Sellers@ usafa.af.mil |
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| MESA Experiment Manager | LtCol Francis Chun<br>719-333-2601, Francis.Chun@usafa.af.mil |
| Principal Investigator | Dr. Linda Krause<br>719-333-4619, Linda.Krause@usafa.af.mil |
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| Structural Engineer<br>(Consultant) | Thomas P. Sarafin of Instar Engineering and Consulting, Inc.<br>303-973-2316, TPSarafin@aol.com |

## 1.4 Safety Analysis

Established analysis methods will be used qualitatively to show the likely hazards associated with the FS-2 subsystems. Operational conditions of the FS-2 payload will be described in detail to identify potential safety hazards and their associated countermeasures, or inhibits, implemented in the design.

## 1.5 Safety Status Summary

The FS-2 design is 100% complete. The FS-2 payload has completed the Phase 0/1 safety review and is preparing for a phase 2/3 safety review.

### *1.5.1 Action Items*

There are no action items at this time pending review by the Payload Safety Review Panel.

### *1.5.2 Noncompliance Reports (NCRs)*

There are no noncompliance reports at this time, nor are any NCRs anticipated for the FS-2 payload.

### *1.5.3 Operational Controls Identification*

None of the hazards identified will need to be controlled operationally.

## 1.6 Applicable Documents

Safety Policy and Requirements for Payloads Using the Space Transportation System, NSTS 1700.7B, NASA/JSC, January 1989.

Payload Verification Requirements, Space Shuttle Program, NSTS 14046E, NASA/JSC, March 2000.

Implementation Procedure for NSTS Payloads System Safety Requirements for Payloads Using the Space Transportation System, NSTS-ISS 13830C, NASA/JSC, July 1998.

Interpretations of NSTS/ISS Payload Safety Requirements, NSTS/ISS 18798, Rev. B, NASA/JSC, September 1997.

Hitchhiker Customer Accommodations & Requirements Specification, 740-SPEC-008, NASA/GSFC 1999.

Guidelines for the Selection of Metallic Materials for Stress Corrosion Cracking Resistance in Sodium Chloride Environment, MSFC-STD-3029, NASA/MSFC, May 22, 2000.

Fracture Control Requirements for Payloads Using the Space Shuttle, NASA-STD-5003, October 7, 1996.

Manned Space Vehicle Battery Safety Handbook, JSC-20793, NASA/JSC, September 1985.

NASA Fastener Integrity Requirements Document, 541-GP-8072.1.2.

Shuttle Orbiter/Cargo Standard Interfaces, ICD-2-19001, rev. L., January 15, 1998.


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# 2. PAYLOAD DESCRIPTION

## 2.1 Mission Objectives

There are three main objectives for the FalconSAT-2 (FS-2) mission:

- The primary science objective is to investigate ionospheric plasma depletions that cause radio transmission disruptions. This will be accomplished using the miniature electrostatic analyzer (MESA) instrument onboard the FalconSAT-2 spacecraft and represents a DoD Space Experiments Review Board (SERB) mission.
- The overall programmatic objective is to provide an opportunity for USAF Academy Cadets to "learn space by doing space," allowing them to participate in all phases of mission design, assembly, test, and operations.
- A final, longer-term program objective is to validate key technologies, design concepts and processes that can be used for follow-on FalconSAT missions.

## 2.2 Physical Description

The primary structure for the FS-2 microsatellite is a 12.5 in (31.75 cm) cube. With antennas and PES adapter ring, the total height is 20.9 in (53.1 cm). The total mass is 43.25 lbs. The external configuration is shown in Figure 1. The majority of FS-2 components will be mounted inside the primary structure. FS-2 subsystems include:

- Structure
- Electrical Power Subsystem (Power Conditioning Unit, Battery, Solar Panels) (EPS)
- Data Handling System
- Communication Subsystem (VHF Rx, S-Band TX) (COMM)
- Thermal Control
- Attitude Determination and Control Subsystem (ADCS)
- MESA Experiment

An expanded view of the spacecraft is shown in Figure 2. Physical characteristics of the subsystems, along with operating temperature ranges are shown in Table 2. The complete functional description for each of the FS-2 subsystems is presented in Section 2.3.

FS-2 will be mounted in a HH Canister (with heating capability), with Hitchhiker Motorized Door Assembly (HMDA) and will use the PES.

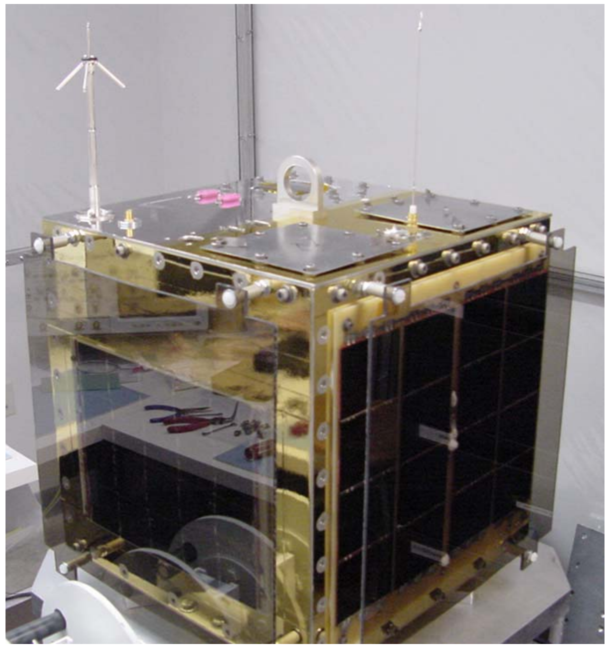

**Figure 1: FS-2 external view**

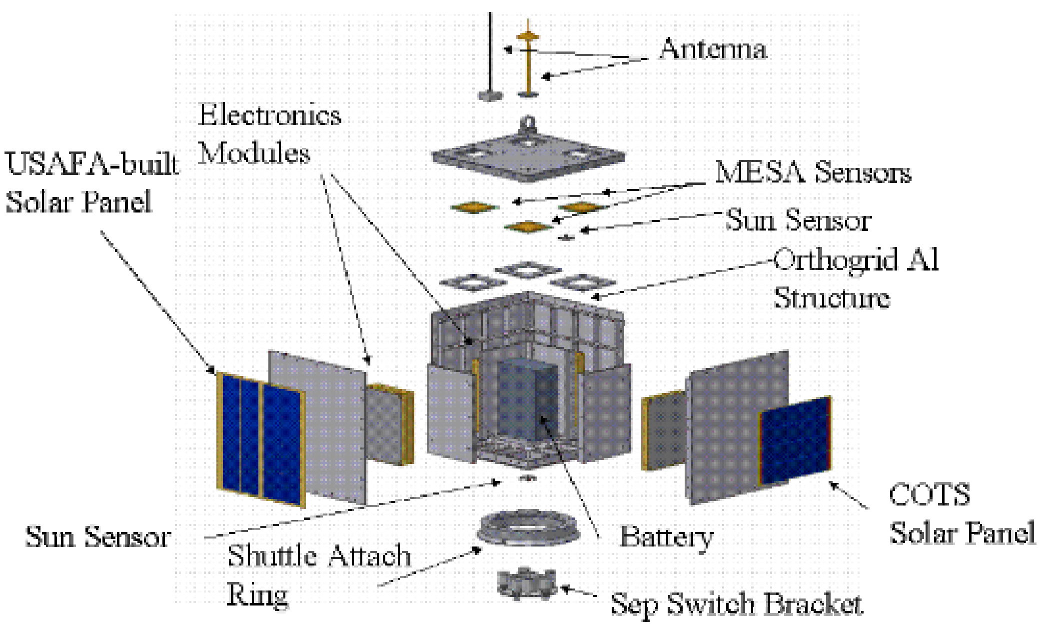


**Figure 2: FS-2 nanosatellite showing key features and components.**

**Table 2: Summary of FS-2 Subsystem Characteristics**

| Item | Payload Characteristic | *MESA* | *Structure* | *EPS* | *Data Handling* | *Comm* | **FS-2** |
|---|---|---|---|---|---|---|---|
| 1 | Weight (approx.) | 1 lbs | 35 lbs | 5 lbs | 2 lbs | 2 lbs | 43.25 lbs |
| 2 | Field of View | 30° | N/A | N/A | N/A | N/A | 30° |
| 3 | Nominal Operating Temp (Min) | 0° C | N/A | 0° C | 0° C | 0° C | 0° C |
| 4 | Nominal Operating Temp (Max) | 30° C | N/A | 30° C | 30° C | 30° C | 30° C |
| 5 | Non-Operating Temp (Min) | -40° C | N/A | -30° C | -40° C | -40° C | -30° C |
| 6 | Non-Operating Temp (Max) | 85° C | N/A | 50° C | 85° C | 85° C | 50° C |
| 7 | Storage Temp (Min) | -40° C | N/A | -30° C | -40° C | -40° C | -30° C |
| 8 | Storage Temp (Max) | 85° C | N/A | 50° C | 85° C | 85° C | 50° C |
| 9 | Safety Temp (Min) | -40° C | N/A | -30° C | -40° C | -40° C | -30° C |
| 10 | Safety Temp (Max) | 85° C | N/A | 100° C | 85° C | 85° C | 50° C |

Note for Table 2
1. Battery is the driver for the min/max temperature requirement.

## 2.3 Subsystem Functional Descriptions

FS-2 hardware is outlined in the following sections:

- Structures (Section 2.3.1)
- Electrical Power System (Section 2.3.2)
- Data Handling (Section 2.3.3)
- Communications (Section 2.3.4)
- Thermal (Section 2.3.5)
- Attitude Determination and Control (ADCS) (Section 2.3.6)
- MESA (Section 2.3.7)

### *2.3.1 Structure*

FS-2 primary structure consists of four milled aluminum side panels, four milled center column plates, a milled base plate, and a milled top panel. All of these components are machined from 6061-T651 and 6061-T6 Aluminum. The spacecraft electronics modules are mounted around the central column walls. The battery box is mounted to the base plate at the center of the column. The top wall will house the sensor arrays, antennas, and a ground handling fixture.

The PES adapter ring is mounted to the base plate and forms half of the PES separation system. The adapter ring was supplied by GSFC. A fit check was performed by GSFC personnel. The base plate will also provide mounting locations for the FS-2 separation micro-switches that will inhibit activation of the spacecraft prior to deployment as well as provide indication of separation from PES. No independent confirmation of separation from the PES is provided by the FS-2 systems.

It is not critical where FS-2 is located in the Payload Bay (PLB). No specific orientation of FS-2 about the PES base plate normal is required.

### *2.3.2 Electrical Power Subsystem*

The FS-2 EPS consists of:

- 4 solar panels
- 1 7-cell battery, housed in a vented aluminum box, vents are along spacecraft y axis
- Power regulation, control and distribution in the form of 2 Printed Circuit Boards (PCB) referred to as the battery charge regulator (BCR), the power conditioning module (PCM) and the power distribution module (PDM). These 2 PCBs are housed together in 2 aluminum trays.

Key features of the FS-2 EPS are summarized in Table 3. A detailed description of each of these components follows.

The solar panels will consist of 2 panels constructed at USAFA and 2 commercial-off-the-shelf panels built by SpaceQuest, Ltd. USA.

All panels will use single-junction GaAs cells. The USAFA-built panels consist of 16 5.5 x 6.5 cm cells bonded directly to printed circuit boards (PCBs) using RTV566 adhesive. The PCBs are then attached to the side panels of the spacecraft structure using fasteners and RTV566.

The Spacequest panels consist of 20 4.1 x 4.1 cm cells bonded to aluminum honeycomb with fiberglass face sheets using RTV566. These solar panels are then mounted to the other two side panels of the spacecraft structure, again using fasteners and RTV566. Photographs of both panels are shown in Figure 3. The components of the solar panels are not considered structural items.

**Table 3: Summary of FS-2 EPS Characteristics**

| EPS Parameter | Value |
|---|---|
| Number of Solar Panels | 4 |
| Orbit Average Power | 2.5 W |
| Bus voltage | 5 V (regulated)<br>8.4-10.5 V (unregulated) |
| Battery Cell Type | N4000DRL |
| Number of Battery Cells | 7 |
| Battery Capacity | 4.3 A-hr |
| Nominal Battery Operating Temperatures | 0°C to 30°C |

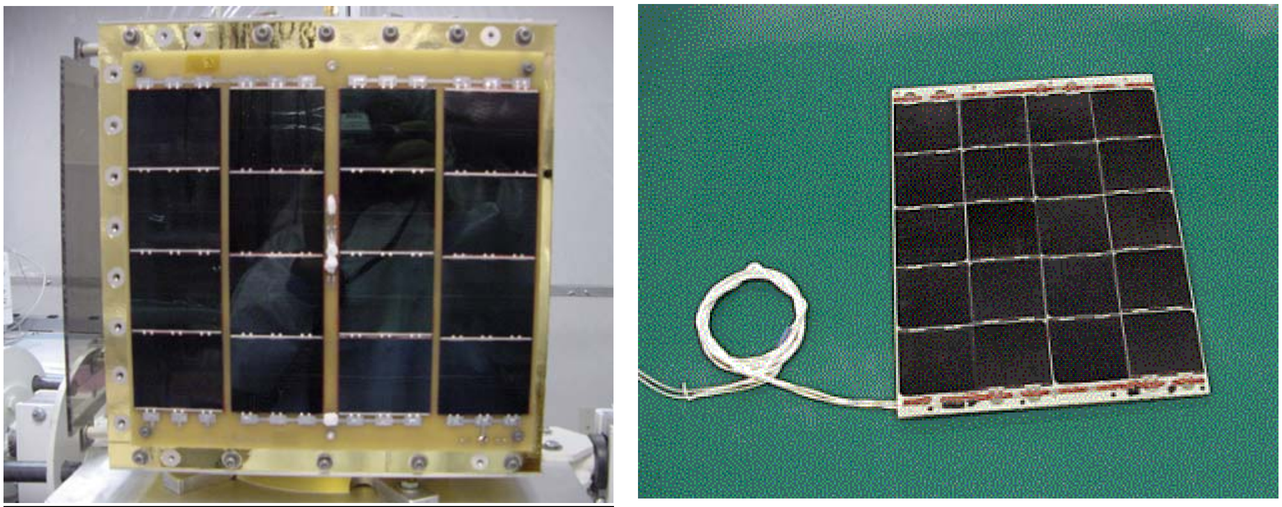

**Figure 3: FS-2 solar panels from two sources.**
A USAFA-built solar panel is shown on the left. A COTS panel from Spacequest is shown on the right.

The FS-2 battery contains 7 Sanyo N4000DRL NiCAD cells. The cells are housed in a milled aluminum box approximately 6 x 4 x 1 inches. The battery was supplied by Surrey Satellite Technology, Ltd. (SSTL). The battery box design and assembly are in accordance with JSC 20793 *Manned Space Vehicle Battery Safety Handbook*. The cell vent is free of obstruction so that the cell is able to vent excessive pressure in the event of an anomaly. For this reason, the ends of the cell were not potted or covered. The battery uses the following design criteria:

- The battery has a slow-blow polyswitch (3 Amp) incorporated into the electrical circuit, into the negative lead of the battery. See Section 10 of the Battery Acceptance Report, located in Appendix E, for details on the slow-blow polyswitch.
- The battery housing completely encloses the cells.
- There are two venting holes on each of the two lids of the battery box. The lids containing the venting holes are oriented along the spacecraft y-axis. The venting holes are covered with PTFE/GORE-TEX disks, allowing air to escape, but containing any electrolyte leakage.
- Pigmat MAT 301 material is used to contain the electrolyte of all the cells.
- The Battery box is leak tight. The use of a Sigaflex graphite gasket between the case and lids provides a leak tight seal, but maintains electrical continuity.
- The internal surfaces of the battery box are non-conductive and resistant to the electrolyte. The external surface of the aluminum battery box is treated with Alodine 1200 and the internal coating is EccoCoat EP3.

Figure 4 shows an external and internal view of the battery box configuration. Figure 5 shows the alignment of the battery box within the spacecraft.

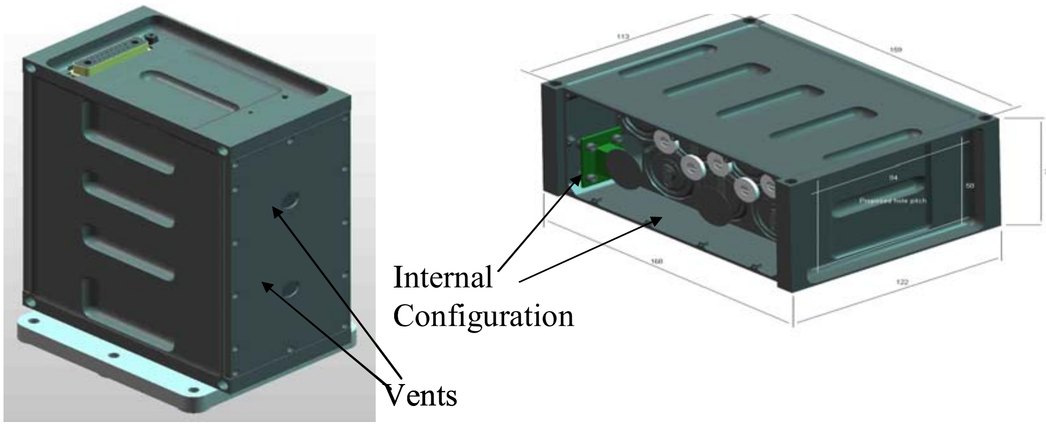


**Figure 4: External and internal views of the FS-2 battery box.**
(battery vent holes are aligned along the spacecraft y-axis)

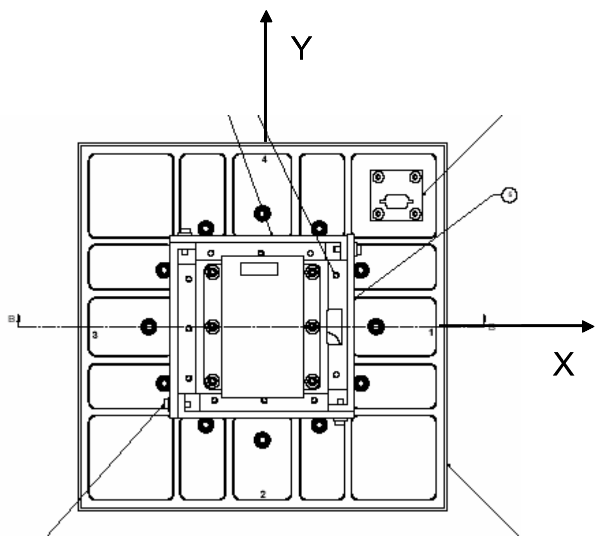


**Figure 5: Battery box alignment.**

Battery charging, power conditioning and power distribution functions are handled by two sets of commercial electronics using surface-mount components on PCBs. There is a separate Battery Charge Regulator (BCR) for each of the four solar panels. The BCRs regulate the voltage and current from the panels to achieve maximum power point and to the battery to prevent over-charging. The Power Conditioning Module (PCM) is basically a DC/DC converter producing a regulated 5V bus voltage line. The Power Distribution Module (PDM) consists of a set of power switches that can be software controlled to turn other spacecraft subsystems on or off.

All spacecraft wiring and polyswitches are rated to meet NASA Shuttle Payload Bay rating requirements (TA-92-038). To prevent premature (prior to release from the PES) battery charging and application of battery power to any system, a two-fault tolerant inhibit scheme is required. As FS-2 will make use of the un-powered bus exception, four independent micro-switches are used to inhibit all battery operations prior to deployment. The function of each independent inhibit will be independently verified on the ground prior to launch. Two inhibits will be on the positive side of the battery, and two on the ground side. The configuration of these switches will be verifiable via a single access port as discussed in Section 3.4.5.

In addition, a single separation switch is installed before the PCM. This switch prevents stray current from the solar panels, prior to deployment, from causing a soft start of the PCM controller, which could cause it to hang up during subsequent post-deployment operation. As this switch is not a Shuttle safety item, no external monitoring of this switch position was developed. The functional schematic for the overall inhibit scheme is shown in Figure 6. Note that that the BCRs, PCM and PDM are physically housed together in two aluminum trays.

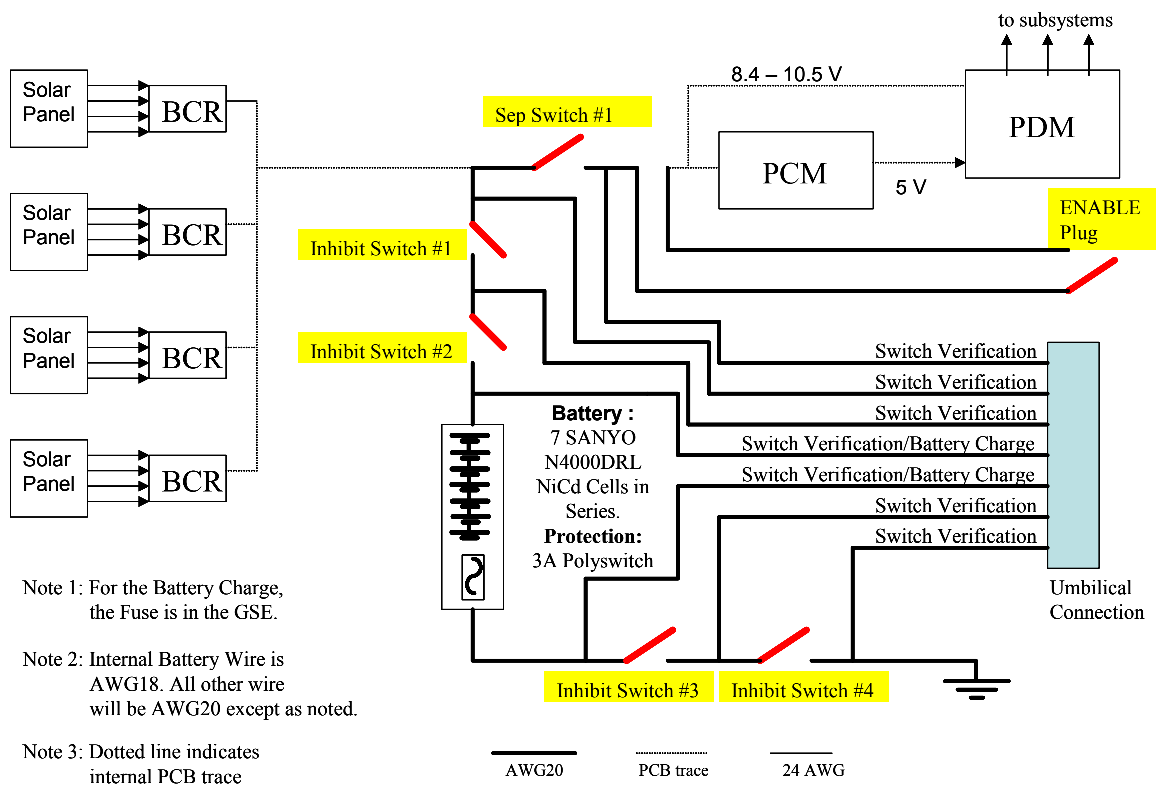


**Figure 6: Battery charge inhibits schematic.**

The FS-2 Battery Charge Regulator (BCR) and Power Conditioning Module(PCM) consist of COTS Printed Circuit Boards (PCBs) supplied by SSTL, UK. The PCBs are housed in milled aluminum trays, which are mounted to the central column walls of the spacecraft. The BCRs provide hardware-controlled maximum power point tracking of the solar arrays via thermistors mounted on each panel. The PCM provides a regulated 5 V current line and an unregulated line at battery voltage (nominally 8.4 V) and includes switches to turn subsystems on/off. Telemetry and commanding of the BCR and PCM is provided via both a control area network (CAN) node as well as through the onboard computer.

### *2.3.3 Data Handling Subsystem*

The FS-2 spacecraft effectively has two independent data handling systems. The default system relies on the CAN. Each subsystem and the MESA experiment has a unique CAN address. The CAN system offers rudimentary built-in telemetry and commanding ability. In addition, a high speed onboard computer (OBC) based on the StrongArm SA1100 processor provides more sophisticated telemetry and commanding capability as well as dedicated support for specific experiment data collection requirements. MESA experiment interface to both the CAN and the OBC will be through a dedicated System Integration Module (SIM). Both the OBC and SIM were purchased from SSTL as COTS PCBs housed in SSTL-standard 6.5 x 4.8 x 0.82 inch aluminum trays (one tray each for OBC and SIM). Detailed descriptions of key data handling subsystem features are provided below.

#### 2.3.3.1 Control Area Network

The CAN system offers rudimentary built-in telemetry and commanding ability. Each subsystem and the MESA experiment has a unique CAN address (EPS, TX, RX, OBC and SIM). The CAN node controller is a MPC5210 microcontroller with serial links to the OBC. The CAN interface conforms to the CAN 2.0A active and 2.0B passive standards. The bit timing on the CAN bus is 2.6 μs (or 385 kbps). The CAN network is daisy-chained through each subsystem module. Each end of the CAN network is terminated with a 120 Ω resistor.

#### 2.3.3.2 Onboard Computer

The FS-2 OBC is based on the high performance StrongArm processor. This processor was chosen because it combines a high-speed reduced instruction set computer (RISC) with low power (up to 220 million instructions per second (MIPS) @ 190 MHz). The StrongArm processor provides interfaces to static, FLASH, and read-only memories (ROM), power and memory management, integrated clock generation, an on-board real-time clock, an interrupt controller, asynchronous communication ports, and general purpose input/output.

The block diagram for the FS-2 OBC is shown in Figure 7. The OBC has a watchdog timer, 1 Mbyte of flash memory, 4 Mbytes of error detection and correction (EDAC) protected random access memory (RAM), an asynchronous Universal Asynchronous Receiver Transmitter (UART) communication link, an interface to a CAN bus, and a synchronous High-level Data Link Controller (HDLC) link.

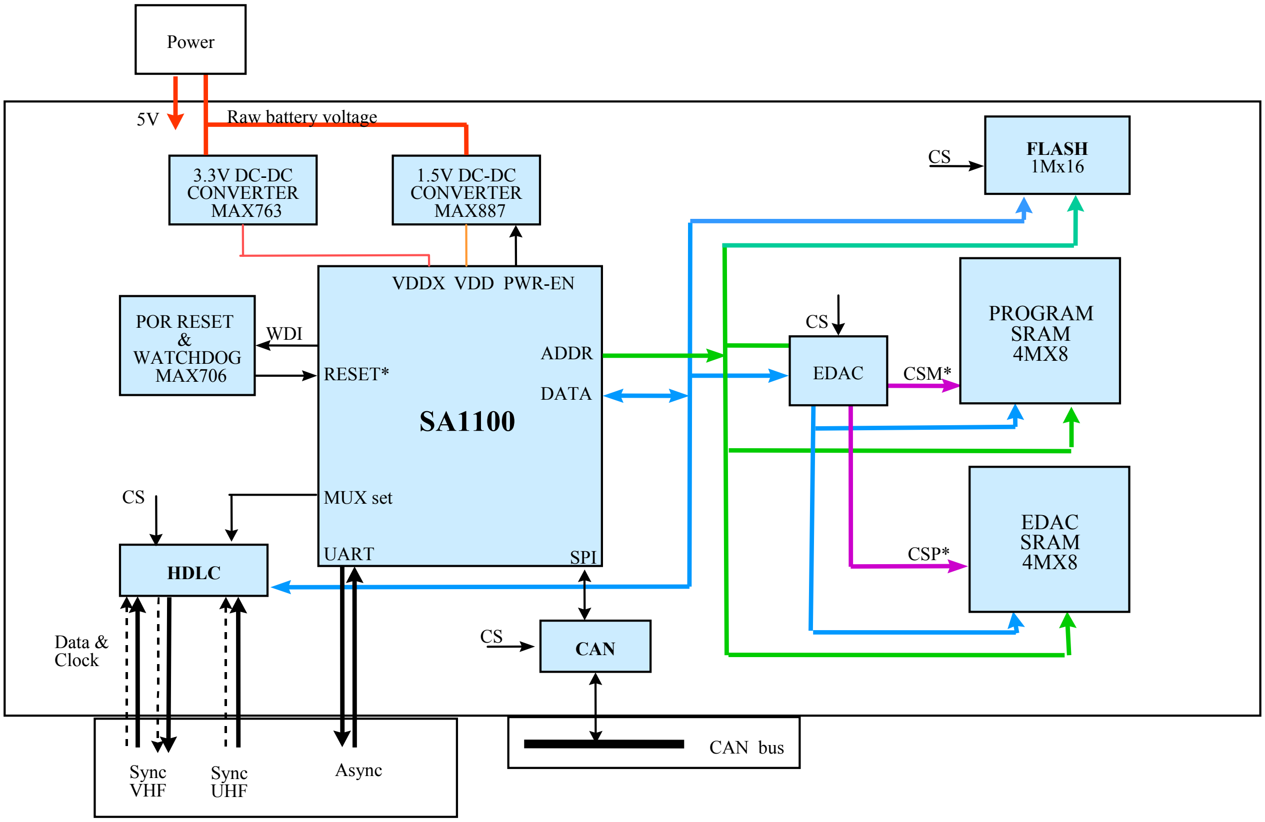


**Figure 7: Functional block diagram for the FS-2 onboard computer subsystem.**

Key features of the FS-2 OBC include:

- Strong Arm SA1100 Processor
  - High performance RISC processor: up to 220 MIPS @ 190 MHz
  - Low power: <330 mW @ 1.5 V/200 MHz
  - Low power: <230 mW @ 1.5 V/133 MHz
  - Power management: normal, idle, and sleep modes
  - Low voltage: 3.3 V/1.8 V
- 1Mx16 FLASH memory
- 4Mx8 EDAC protected program memory
- WATCHDOG Timer
- CAN bus
- Asynchronous communication link, (uplink at 9.6k baud / downlink at 38.4k baud)
- Synchronous communication link, programmable in a Field Programmable Gate Array (FPGA)
- PCB size: 100 mm x 160 mm, populated on both sides

The FS-2 OBC is designed around the SA1100-EA microprocessor capable of operating at frequencies up to 190 MHz. The CPU clock can be provided by an external clock circuit or by connecting the SA1100 clock inputs with two external crystals of 3.68 MHz and 32.8 kHz. The latter method was chosen. An internal phase-lock loop generates the required internal clock frequency from the 3.68 MHz crystal while the 32.8 kHz crystal is used as a timebase for the real-time clock. To improve stability two 33 pF capacitors connect the 3.68 MHz crystal to ground.

#### 2.3.3.3 Software

Software control of the FS-2 spacecraft via the OBC will accomplished using a single, compiled program written in ASCII standard C. Primary software tasks include:

- MESA sensor control and data gathering
- Orbit propagation for timing of onboard commanded events
- Telemetry collection, real-time downlink and storage
- High-speed automatic data download

Operationally, all mission software will be uplinked as part of spacecraft commissioning. Basic mission software is loaded into the OBC flash memory and will be available on boot-up of the OBC. However, it is anticipated that upgrades to this basic software will be uplinked to improve performance over the lifetime of the mission.

#### 2.3.3.4 Subsystem Integration Module (SIM)

The SIM provides primary data interface between the MESA experiment and the spacecraft bus. The SIM has two connectors, a D-9 for power and CAN bus connections and a D-44 high density for connections to MESA and sun sensors. The signals provided on the payload connector include:

- Eight analog inputs (range 0 - 4.1 V) sampled using a ten bit A/D converter
- Eight analog outputs (range 0 - 4.1 V) generated with eight bit D/A converters
- Sixteen bi-directional digital lines (open drain)
- Asynchronous serial port (TTL signaling)
- Analog reference voltage (4.1 V)
- Power supplies (5 V and Vbatt)

A functional block diagram for the SIM is shown in Figure 8.


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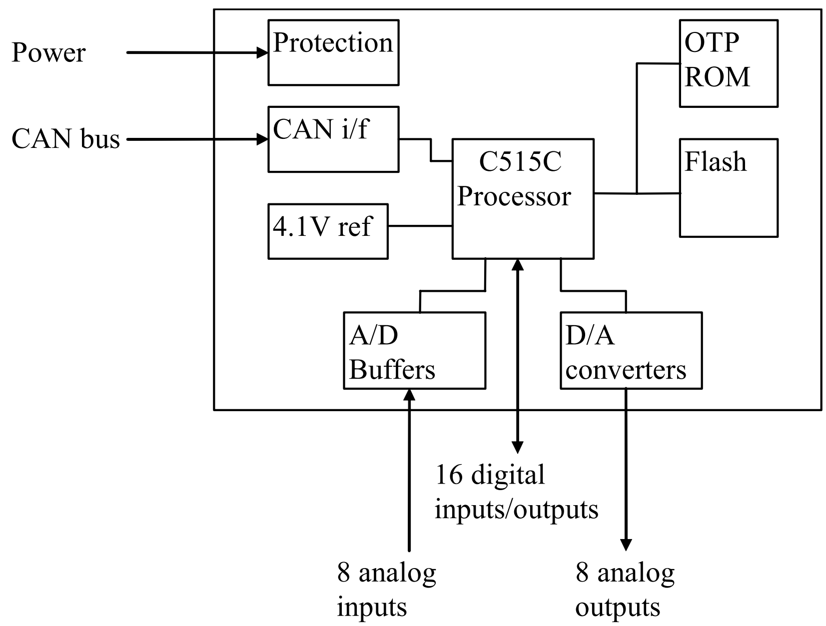


**Figure 8: Functional block diagram for the FS-2 SIM.**

### *2.3.4 Communication Subsystems*

The FS-2 communication subsystems consist of an S-band transmitter (TX) and a VHF receiver (RX). Both subsystems were purchased from SSTL as COTS PCBs housed in SSTL-standard 6.5 x 4.8 x 0.82 inch aluminum trays (one tray each for the TX and RX). Detailed descriptions of each subsystem are provided below.

#### 2.3.4.1 S-band transmitter

The S-band transmitter will operate at a nominal frequency of 2220.0 MHz at an output power of 500 mW. Power consumption is 4.9 W when the two RF2126 amplifiers are on and 1.5 W when they are powered down. A single S-band ground plane antenna is mounted to the outside top panel of the spacecraft structure and has dimensions of 6.6 inches in height and 2.2 inches in maximum diameter. Figure 9 shows the S-band antenna mounted on FS-2.

The TX employs a CAN controller and interface, and data are sent to the transmitter via the CAN bus or direct from the OBC via the D44 connector. The Downlink modulation scheme uses binary phase shift keying (BPSK) at 38.4k bps at 2.22 GHz. The Transmitter also uses a field programmable gate array (FPGA) to implement the encoders and scrambler, and the ‘I’ and ‘Q’ generation.

S-Band Antenna VHF Antenna

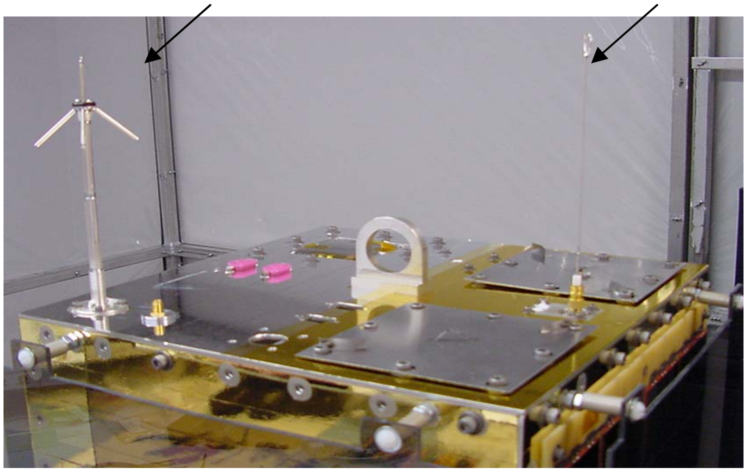

**Figure 9: Location of FS-2 VHF and S-band antennas.**

A functional block diagram for the S-band TX is shown in Figure 10. Key features of the TX include:

- 38.4k bps @ 2.22 GHz
- BPSK
- 500 mW output power
- CAN Bus interface
- PCB size: 100 mm x 160 mm

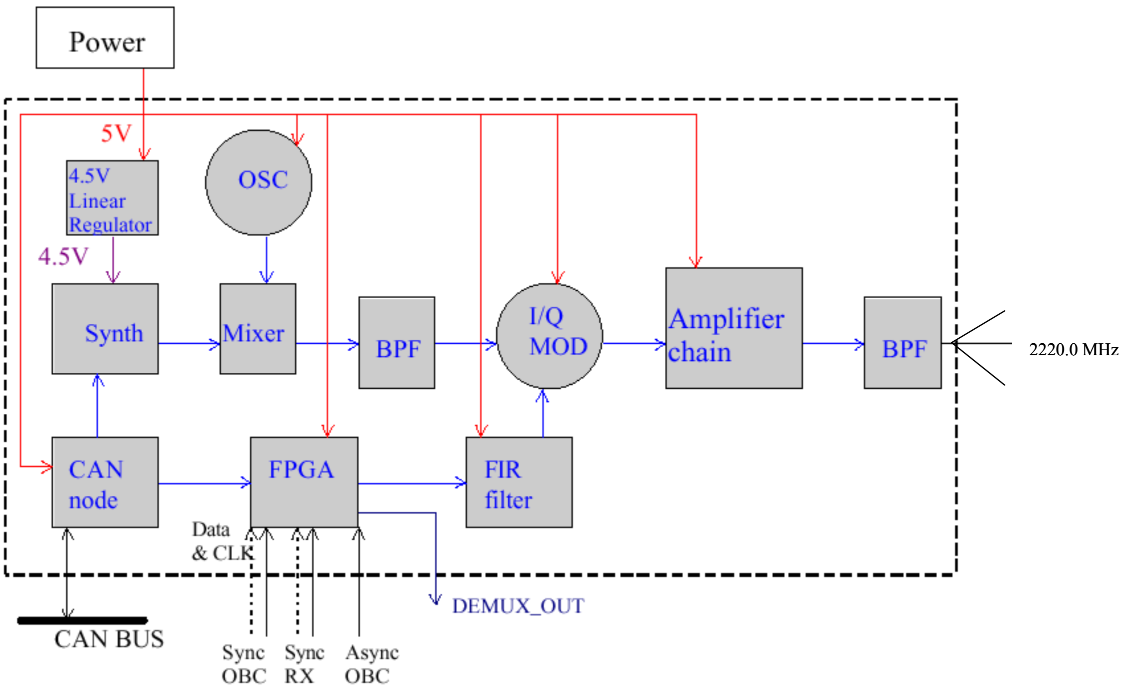


**Figure 10: Functional block diagram for the FS-2 S-band transmitter subsystem.**

Analysis of the radiated RF power from the S-band transmitter indicates it will be below the payload-to-orbiter limit specified in (ICD-2-19001) with the payload bay doors open.

With the doors closed, four inhibits prevent battery power from being inadvertently applied to the TX. There is no safety concern with the possibility of powering the TX directly from the solar panels. As the analysis described in section 5.2.5 shows, the RF energy is below the limit with PLB doors open; and of course, the solar panels can only generate power in this case.

#### 2.3.4.2 VHF Receiver

A regulated 5 V bus power line powers the VHF Receiver module, and the RX section itself runs off an internally regulated 4 V line. The power consumption is approximately 400 mW. A VHF monopole whip antenna is used, mounted to the outside top panel of the spacecraft as shown in Figure 9 extending 6.9 inches above the spacecraft surface. The front end of the RX consists of a low-pass filter (LPF), a 20 MHz band-pass filter (BPF) and a low-noise amplifier (LNA) circuit. The RX employs a CAN controller and interface, and recovered data can be distributed via the CAN bus or direct to the OBC via the D44 connector. The uplink modulation scheme used is 9600 bps frequency shift keying (FSK) at 148.290 MHz.

Key features of the FS-2 RX include:

- 9.6k bps FSK @ 148.290 MHz
- Low power: 400 mW @ 5 V
- 1$^{st}$ intermediate frequency (IF): 21.4 MHz with matched pair of filters
- CAN Bus interface
- Board size: 100 mm x 160 mm
- Voltage buffered receiver signal strength indicator: 70 dB usable range
- Local Oscillator (L.O.) suppression: 40 – 60 dB

A functional block diagram for the FS-2 RX is shown in Figure 11.

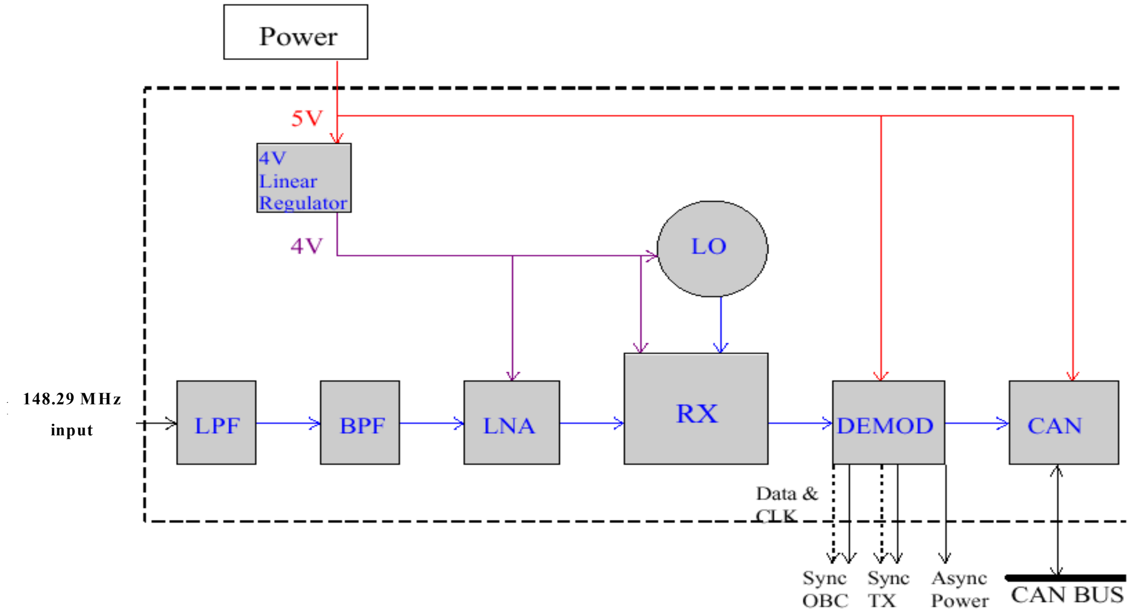


**Figure 11: Functional block diagram for the FS-2 VHF receiver.**

### *2.3.5 Thermal Control*

The thermal control subsystem's purpose is to maintain the temperature of the spacecraft components within the desired temperature limits specified by the component manufacturer. Additionally, the thermal control subsystem seeks to minimize the thermal cycling, or fluctuation, about the equilibrium temperature of each component. On FalconSAT-2, we are using a passive thermal control subsystem consisting of thermal tapes applied to the exterior of the spacecraft.

Because the baseline thermal behavior of FalconSAT-2 without any thermal control applied is already within temperature limits, our thermal tape design does not need to modify the thermo-optical properties of the structure greatly. However, another consideration in determining the thermal design is the attitude control requirement of inducing a spin to ensure that the solar panels receive sunlight. To do this, thermal tapes with differing absorptivity values will be adhered to each side in a pattern that will induce a spin due to torque created by solar radiation pressure.

FalconSAT-2 thermal control requirements can be summarized as follows:

- Maintain all components within operational limits—The operational temperatures are limited by the electronic components within the satellite, and specifically by the battery. The battery is the most thermally sensitive of the satellite subsystems. As a result, the nominal temperature range targeted for the batteries and internal components of FalconSat-2 is 0 to +30 deg C. The other commercial electronics within the satellite have temperature limits of -40 and +85 deg C. The structural components and solar panels have much more relaxed temperature limits. Table 2 summarizes the temperature limits for FalconSat-2.

- Passive attitude control—Attitude control requirement that must considered in the thermal design. Differential thermal coatings will be used to impart a spin due to solar radiation pressure torque on the satellite to ensure that the solar panels are always receiving sunlight, and that no one side is constantly pointed at the sun.

After running simulations using the thermo-optical properties of several combinations of thermal tape, we decided to use a combination of aluminum, Kapton, and gold thermal tapes. Specifically, the tapes shown in Table 4 will be used.

**Table 4: FalconSAT-2 Thermal Tapes**

| TAPE | NAME | ABSORPTIVITY | EMMISSIVITY |
|---|---|---|---|
| Aluminum | Sheldahl Second Surface Aluminum Polyimide Tape with 966 Acrylic Adhesive (Item # 146520) | 0.14 | 0.05 |
| Kapton | Sheldahl First Surface Aluminized Polyimide Tape with 966 Acrylic Adhesive (Item # 146385) | 0.39 | 0.63 |
| Gold | Sheldahl First Surface Gold Coated Polyimide tape with 966 Acrylic Adhesive (Item # 146482 | 0.20 | 0.02 |

The thermal tapes were placed on the satellite in a configuration that will induce a torque via solar radiation pressure on the top and bottom of the spacecraft. On the +Z and –Z facets, which are the facets without any solar panels, half of the surface was covered with aluminum tape and half with Kapton tape. This is because our primary objective on these sides is to induce a torque from solar radiation pressure to avoid a situation with no solar panels illuminated by the sun. For the +X, -X, +Y, and –Y facets, which are the facets mounted with solar panels, only gold tape was used. This is because modeling and simulation shows that the satellite tends to run cold, therefore, the maximum a/e ratio tape should be on most sides, while maintaining the solar radiation pressure torque on the +/- Z facets of the satellite. A diagram of the thermal tape design for each facet of the satellite is shown in Figure 12.


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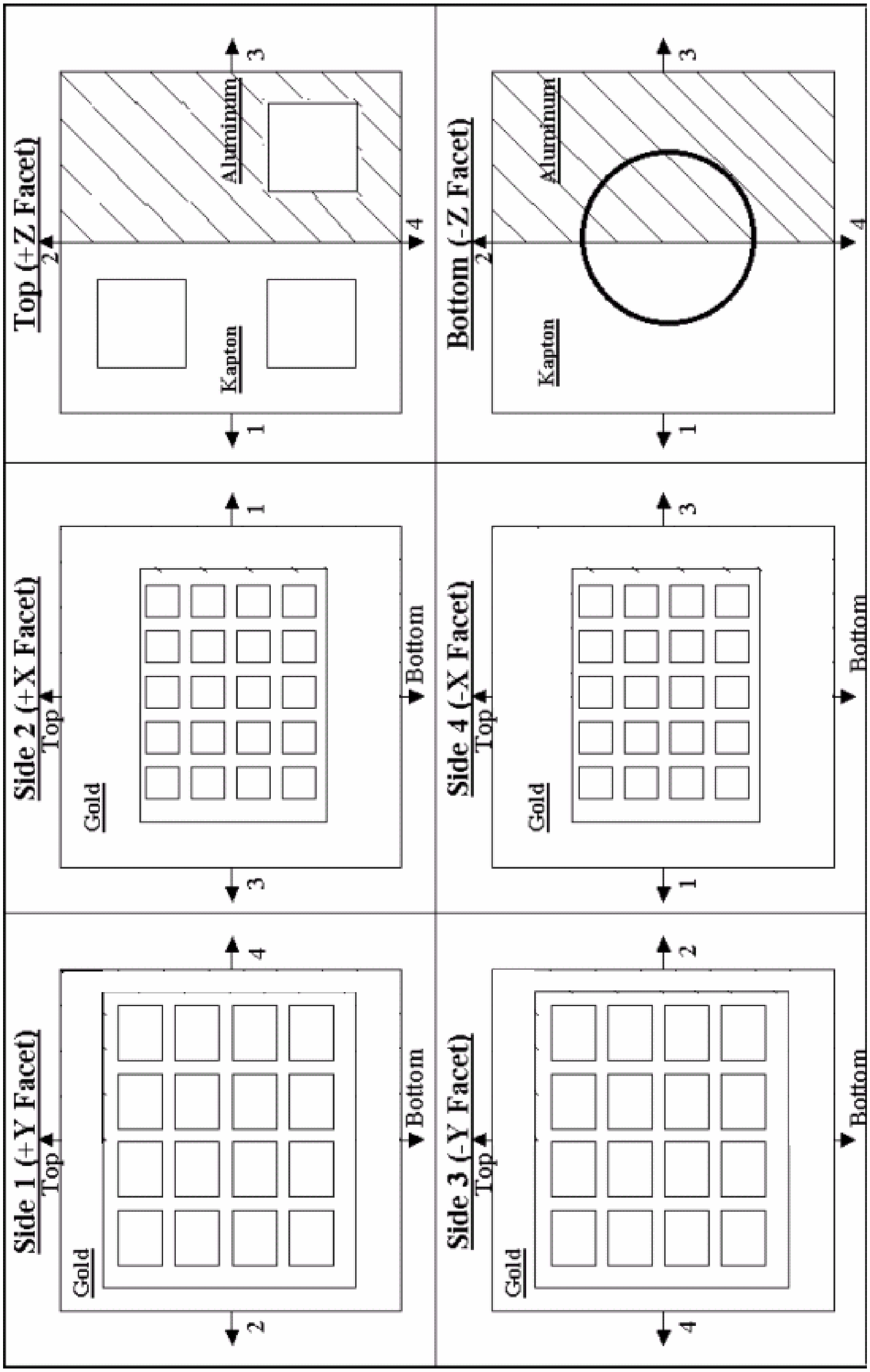


**Figure 12: FalconSAT-2 Thermal Tape configuration.**

### 2.3.6 Attitude Determination & Control (ADCS)

FS-2 will use a combination of Sun sensors and solar panel current to determine spacecraft attitude. One Sun sensor is mounted on the top panel of the spacecraft on the same face as the antenna pointed in the spacecraft +Z body direction. A second Earth/Sun sensor is mounted on the base plate pointed in the –Z body direction. Figure 13 shows the FS-2 body axis coordinate system. The Sun sensor is not shown. Each sensor will use a single photo-resistor with an effective field of view of approximately 100°. Spacecraft attitude will be determined post hoc using ground processing of telemetry.

Passive attitude control will be achieved through using atmospheric drag and solar-pressure spin tapes. The spin tapes will provide both passive thermal control, as discussed in section 2.3.5, and provide areas of high or low solar absorptivity. This differential absorptivity is predicted to be sufficient for solar pressure to induce a slight spin. This passive control scheme is designed to ensure the spacecraft does not become inertially locked in an unfavorable attitude such that the solar panels are not pointed at the sun for at least part of each orbit.

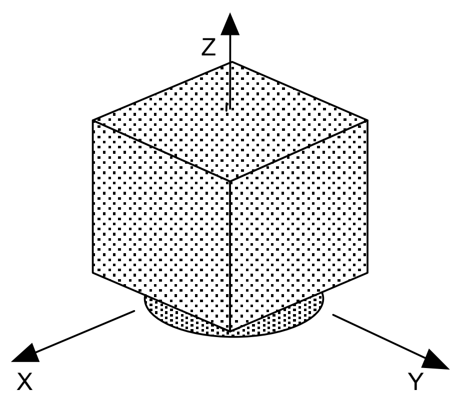


**Figure 13: FalconSAT-2 body axis coordinate system.**
+Z is out the top plate of the spacecraft. –Z is out through the interface ring.

### 2.3.7 Miniature Electrostatic Analyzer (MESA)

The primary science objective of the FS-2 mission is operation of the MESA experiment. MESA will not be activated until after FS-2 is deployed from the Shuttle and complete spacecraft checkout and commissioning. It is inhibited from doing so by the 4 inhibits placed around the FS-2 battery as described in Section 2.3.2. An overview of the MESA instrument is presented here to highlight that there are no Shuttle safety issues associated with the presence of MESA. A discussion of MESA science objectives and operations is included here for information only.

The MESA experiment is designed to investigate low latitude ionospheric plasma depletions with a novel, miniaturized electrostatic analyzer. The MESA experiment is an electrostatic analyzer in the form of a patch sensor designed to measure electron fluxes of energies from thermal up to 100 eV. The MESA experiment consists of two laminated analyzers (LA) and a single electron retarding potential analyzer (RPA) that will operate as a suite to provide *in situ* sampling of ionospheric electron density and temperature along the FS-2 orbit track. The MESA experiment will sample the electron density over 6 energy channels and at a rate of 10 spectra per second. The two LAs and the RPA are mounted to the top panel of the spacecraft as shown in Figure 14.

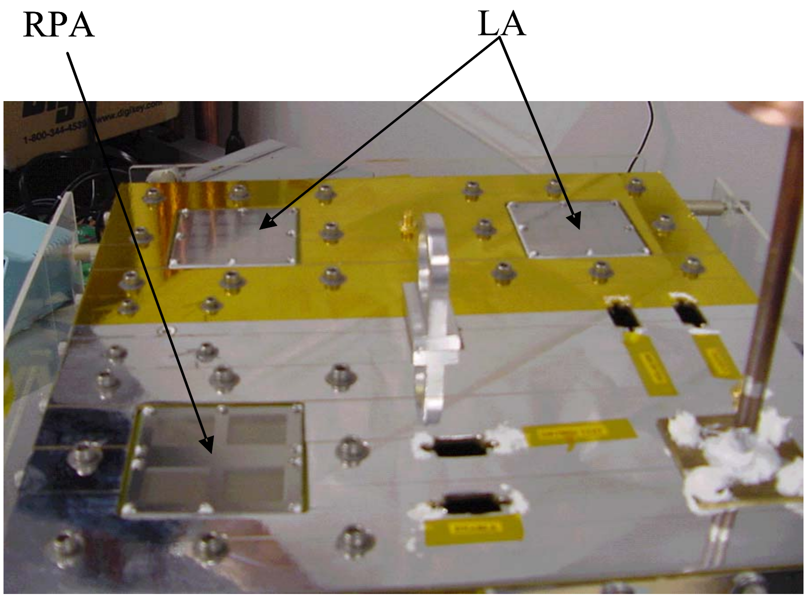


**Figure 14: Arrangement of the MESA patch sensors on the top panel of FS-2.**

#### 2.3.7.1 Engineering Layout

Both the two LAs and the RPA have the same external dimensions: 3.2 in. x 3.2 in. x 0.40 in. Refer to Figure 15 for the top and side views of the LA. Each sensor consists of a single printed circuit board (PCB) with surface-mount electrical components. A stack of thin stainless steel perforated plates separated by alumina spacers are then attached to the PCB to act as “lenses” to separate out electrons of various energies. The MESA sensors will be mounted such that the stainless steel plate is flush with the spacecraft top surface leaving the PCB inside the spacecraft. A configuration diagram is shown in Figure 16. Mounting to the structure is via 4 fasteners through a bracket on the inside of the spacecraft. Thus, the only exposed portions of MESA are the stainless steel plates. Therefore, there is no risk of shattering the MESA sensors due to kick loads.

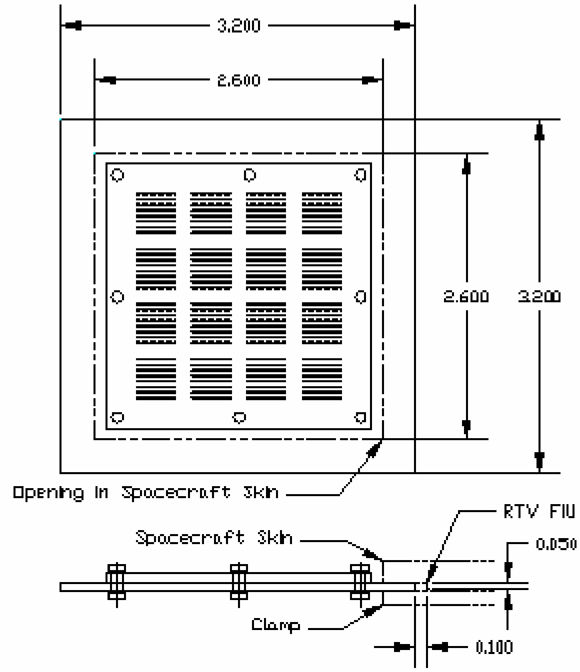


**Figure 15: MESA mechanical drawings, top and side views.**

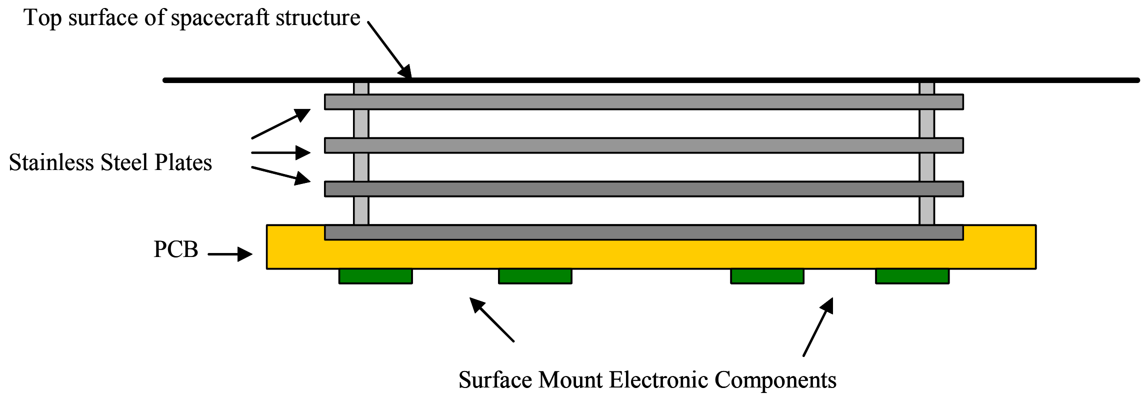


**Figure 16: Physical layout of MESA sensor**

2.3.7.2 Scientific Objectives

The primary (minimum) objectives of the MESA experiment include: (1) To investigate the morphology of plasma depletions in the F region ionosphere, especially in the low and mid latitudes, and (2) to demonstrate the utility of MESA in the measurement of thermal ionospheric electrons.

The secondary (desired) objective is to investigate the structure and evolution of ionospheric plasma bubbles by taking advantage of opportunities to make *in situ* multi-point measurements of electron densities within a single structure. To accomplish this objective, the MESA experimenters will coordinate a data exchange with experimenters from other Low Earth Orbit (LEO) missions making *in situ* plasma measurements in the ionosphere.

Software control of the MESA sensors is initiated through the Subsystem Integration Module (SIM) and a dedicated MESA Interface Board (MIB). Together, these two PCBs manage all data interface to/from the sensors and OBC. No MESA operations will be initiated until the spacecraft is fully commissioned following deployment.

2.3.7.3 Operations Concept

The primary operating mode of the MESA sensors is the data collection mode during spacecraft eclipse. The MESA data collection will begin 50 km (+/- 35 km) before crossing from the dayside into the penumbra of the Earth. Collection should continue until the spacecraft is exposed to sunlight (*i.e.*, when the spacecraft exits the Earth's penumbra heading into the dayside ionosphere). MESA requires that an average of 15 minutes of eclipse time per orbit should be spent collecting data at magnetic latitudes below 45°.

In the fast data collection mode, the MESA sensors collect 10 spectra of electron fluxes per second. With a 16 bit data word, a 6 channel data collection mode would result in 16 bits × 6 energy channels × 10 spectra per second = 960 bits per second. Since the MESA experiment collects data during eclipse, for an average eclipse time of 45 minutes per orbit, the nominal data collection rate per orbit is 2.59 Mbits/orbit. Data reduction algorithms will be employed to reduce the downlink requirement to a nominal value of 200 kbits/orbit, not to exceed (NTE) 256 kbits/orbit. A slow data collection mode (1 spectrum per second) will be available for contingency purposes.

2.3.7.4 Experiment Success Criteria

The MESA experiment success criteria has been established for each experiment objective (refer to sub-section 2.3.7 introduction). They are as follows:

- Primary Objective 1: The level of success is the fraction of the 6 month minimum experiment duration during which data in fast mode (*i.e.*, 10 spectra per second) were obtained, reduced, and successfully downlinked. For example, if MESA gathered data in fast mode over 4 months only, then the mission would be 67% successful in attaining this particular objective. If the instrument had to be placed in slow mode (*i.e.*, 1 spectra per second,) for greater than 50% of the nominal 6-month mission duration of data collection, then the success rate should be halved.
- Primary Objective 2: The level of success is a) 100% if it is demonstrated that MESA is capable of measuring thermal ionospheric electrons over 6 energy channels at a rate of 10 spectra per second, b) 75% if it is demonstrated that MESA is not capable of making these measurements at 10 spectra per second, but it is demonstrated that samples of 1 (6-channel) spectra per second are feasible and reliable, c) 75% if it is demonstrated that MESA is capable of measuring bulk electron flux to the instrument at a rate of 10 samples per second, but is not capable of resolving the spectra into energy channels, and d) 50% if it is demonstrated that MESA is capable of measuring bulk electron flux to the instrument at a rate of 1 sample per second, but is not capable of resolving the spectra into energy channels.
- Secondary Objective: Success is 100% if MESA passes through a plasma depletion simultaneously with another satellite, with both spacecraft making *in situ* ionospheric plasma measurements. Success is 100% even if there is only one encounter of this type during the entire mission. Success is less when the two satellites enter the plasma depletion at different times: Success = $[1-(\Delta t)/t_{orbit}] * 100\%$, where $\Delta t$ is the difference between the time the first satellite exits the plasma depletion and the time the second satellite enters the same plasma depletion, and $t_{orbit}$ is the average orbital period of the

two satellites. Since MESA has no control over the orbit of the second satellite, it would be beneficial to select a spacecraft precession rate that maximizes the number of conjunctions with a second, well known ionospheric plasma diagnostic spacecraft (*e.g.,* Air Force Research Laboratory's Communication/Navigation Outage Forecasting System (C/NOFS)) during eclipse and near the magnetic equator.

Minimum MESA experiment success is attained if the average success level of the two primary objectives is greater than 75%.

Desired mission success is attained when the following two conditions are met: 1) the average success level of the two primary objectives is greater than 90%, and 2) the secondary objective success level is greater than 50%.

2.3.7.5 Experiment data downlink

The MESA experiment has neither its own data storage capacity, nor its own transmitter. The experiment will generate approximately 256 kbits/orbit with the application of data reduction algorithms employed by the FS-2 flight computer.

There are no hard requirements on the download times. MESA data will be time-stamped with spacecraft Mission Elapsed Time (MET). Spacecraft latitude, longitude, magnetic local time, and Universal Time, all as a function of MET, will be used for post-processing.

2.3.7.6 Other Environmental Requirements

The MESA operating and storage temperatures are specified in Table 2. MESA has no requirements regarding thermal isolation, humidity, or atmospheric pressure. There are no safety concerns with exceeding these temperatures, however proper functionality of the MESA experiment will not be ensured if exceeded.

## 2.4 Operations

Deployment and operation of FS-2 hardware occurs in five phases as shown in as shown in Table 5.

**Table 5: FS-2 Operational Timeline**

| Phase | Event | Mission Time | Power Status |
|---|---|---|---|
| 0 | Awaiting deployment | <T0 | Not powered |
| 1 | FS-2 ejection from PES, release of micro-switches removes electrical inhibits to EPS, receiver turned on (if sufficient solar and/or battery power available) | T0 | Powered |
| 2 | Battery charging | T0 to T+10 hrs | Powered |
| 3 | First contact, transmitter turned on by ground command | First pass over USAFA >T+10 hrs | Powered |
| 4 | Begin commissioning | Subsequent passes over USAFA | Powered |

At T0, ejection of FS-2 from the PES, all the micro-switches that inhibit the EPS will close. This will allow power to flow from the solar panels to the battery and the PCM. The mission operations concept calls for a minimum of 10 hours to elapse (approximately 6 orbits) to allow full charging of the batteries before first contact with the satellite is initiated from the USAFA ground station. With power available onboard, only the EPS and RX will automatically be powered on. During first contact, the TX will be commanded on by ground controllers. Following initial contact, spacecraft commissioning will begin. No NASA operational assistance is required after release.

#### *2.4.1 Critical Procedures*

No on-orbit critical procedures have been identified for FS-2.

## 3. PAYLOAD REQUIREMENTS FOR STANDARD CARRIER SERVICES

### 3.1 Carrier to Payload Electrical Interfaces

FS-2 will require no carrier to payload electrical interface connections.

### 3.2 Carrier to Payload Mechanical Interfaces

FS-2 is attached to the Shuttle Hitchhiker Pallet Ejection System (PES) via an adapter that is mounted to the bottom surface of the spacecraft. The FS-2/PES will be installed in a canister with lid. The entire assembly is installed in the Shuttle Orbiter Payload Bay. The FS-2 mechanical interfaces meet the standard interface specifications for PES.

### 3.3 Carrier to Payload Thermal Interfaces

The payload meets the standard thermal interface requirements specified in the CARS for PES payload in a canister with HMDA and heaters. The FS-2 thermal design will rely exclusively on passive thermal control consisting of judiciously selected surface coatings post-deployment.

### 3.4 Ground Operations Requirements

This section outlines the plan for FS-2 ground operations at GSFC and the Kennedy Space Center (KSC). Ground operations consist of physical examinations, functional & verification testing, servicing, and final flight preparation, including removal of Remove Before Flight (RBF) covers, see Table 6. It is requested that all activities be accommodated at both GSFC and KSC, except for final flight preparations, which are conducted only at KSC. Areas at KSC at which the FS-2 may be accessed include the Payload Processing Facility (PPF), the Orbiter Processing Facility (OPF), and the launch pad (PAD).

**Table 6: Ground Operations Locations**

| Ground Operation | GSFC | KSC | Note | Ref Section |
|---|---|---|---|---|
| Physical Examination | YES | | Inspection of fastener torques and integrity of separation systems | 3.4.1 |
| Pre-PES Installation | YES | | Removal of RBF covers, Insert and secure ENABLE plug, Battery charging | 3.4.2 |
| Functional Testing | YES | PPF | RF transmission and reception | 3.4.3 |
| Verification Testing | YES | PPF | Safety inhibit verification | 3.4.3 |
| Servicing | YES | PPF | Remove solar panel covers, Battery top-charging, Battery conditioning | 3.4.4 |
| Final Flight Preparations | | PPF, OPF | Battery top-charging, removal of RBF covers | 3.4.5 |
| On-PAD Battery charging | | PAD | Battery top-charging | |

Note for Table 6.

1. Only PAD requirement is for battery charging on a non-interference basis only if time since final PPF top-off charge exceeds 45 days (see Section 3.4.4).

### *3.4.1 Physical Inspection (GSFC/KSC)*

At GSFC site, FS-2 will be subjected to a post delivery inspection to determine the payload's physical condition following transport, handling, and the passage of time. Physical examination involves checking the exterior surface for scratches, dents, and sharp edges; breakage of solar cells; fastener torques; and separation systems mechanical integrity.

### *3.4.2 Pre-PES Installation*

Prior to installation of FS-2 on to PES, several operations will need to be performed:

- Top charging of the battery (depending on elapse time since previous charging)
- Removal of all RBF covers
- Insertion and securing of the ENABLE plug

Three RBF sensor covers and four RBF solar panel covers must be removed as shown in Figure 17. A summary of all RBF and IBF items is detailed in Table 7.

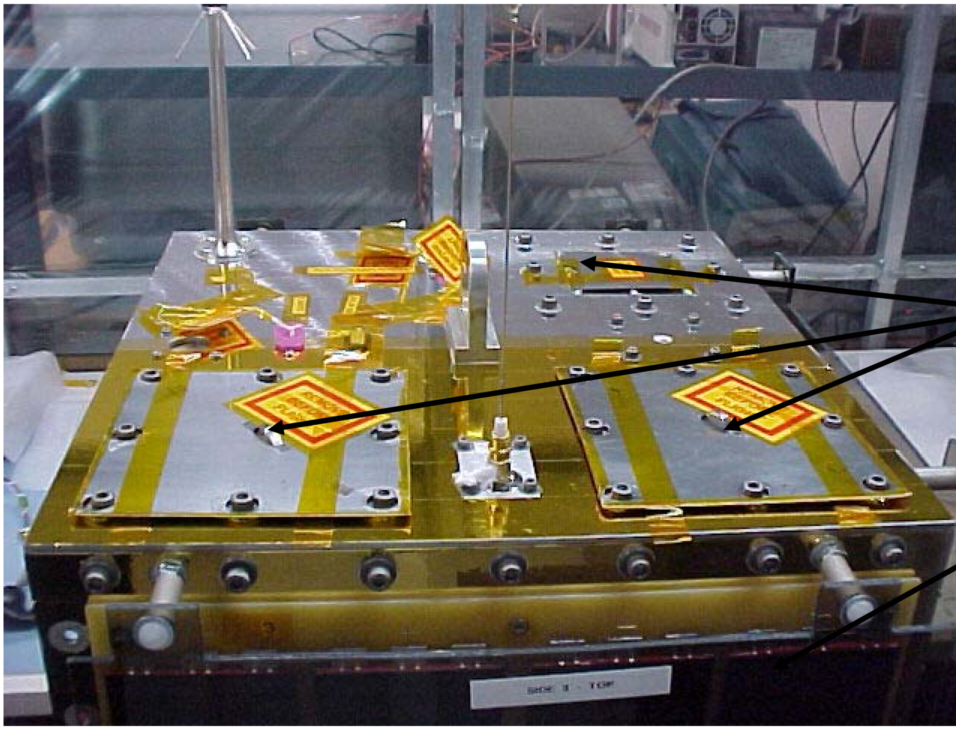


**Figure 17. Locations of RBF/IBF Covers**

**Table 7: FS-2 Locations of RBF/IBF Items**

| Name | Type | Physical Location | Description | When removed/installed |
|---|---|---|---|---|
| Solar Panel Covers | RBF | Over each solar panel (4 total) (see Figure 17) | Plexiglass covers secured to the side of the spacecraft with fiberglass fasteners during transport and handling | Prior to installation in the PES container |
| MESA/RPA Covers | RBF | Over each MESA/RPA sensor (3 total) (see Figure 17) | Aluminum covers secured to the top of each sensor with kapton tape during transport and handling | Prior to final HH can closeout |
| ENABLE plug | IBF | Top Panel (see Figure 18) | Female D9 connector with pin-to-pin wiring to make appropriate connections within the spacecraft circuitry | Prior to installion of the spacecraft into the PES |
| CAN/TTL Port Cover | IBF | Top Panel (see Figure 18) | Plastic D9 connector cover secured with RTV | Prior to final spacecraft closeout |

| | | | | |
|---|---|---|---|---|
| Solar Simulator Power Port Cover | IBF | Top Panel (see Figure 18) | Plastic D9 connector cover secured with RTV | Prior to final spacecraft closeout |
| Switch Test/Battery Charging Port Cover | IBF | Top Panel (see Figure 18) | Plastic D9 connector cover secured with RTV | Prior to final spacecraft closeout (may have to be removed and re-installed to support battery charging on the PAD as necessary) |

The ENABLE plug must be installed during pre-PES installation to allow FS-2 circuitry to operate as intended. The ENABLE plug acts as an additional spacecraft functional inhibit for ground testing and handling purposes. The ENABLE plug must be installed before flight (IBF) into the ENABLE port, which is a male DB-9 connector on the top panel of the satellite as shown in Figure 18. The ENABLE plug consists of a female DB-9 connector containing the enable wiring as shown in Figure 19. This plug simply shorts the appropriate pins to allow the spacecraft to function. There are small hold-down screws as part of the ENABLE plug which will be tightened and secured with RTV for flight.

The status of all inhibits will be verified as necessary through the Switch Test and Battery Charge port, which is a female DB-9 connector as shown in Figure 18. This verification process utilizes a temporary connection.

### *3.4.3 Functional and Verification Tests (GSFC/KSC)*

Functional and verification tests will be required to identify any FS-2 functions that may have been affected by transport, handling, or the passage of time.

#### 3.4.3.1 Functional Testing

FS-2 functional testing will include supplying external bus power to activate all electronic components, after which each subsystem will be methodically tested according to a documented set of procedures (note: external bus power will be supplied through ports as shown in Figure 18. These functional tests will require a power supply, laptop computers, and portable GSE. FS-2 will furnish this equipment as well as any specialized cables and wiring.

#### 3.4.3.2 Verification Testing.

Verification testing includes testing of safety inhibits to verify their status. Verification testing will be conducted through a port at the top surface of the payload as shown in Figure 18. The FS-2 program shall provide equipment and procedures for verification testing. Any inhibits removed during testing will later be enabled as part of the test procedure and later re-verified.

In addition to verification testing of safety inhibits, EMI/EMC testing will be conducted at GSFC (by NASA personnel) to test effects of radiated emissions of the type expected in the Orbiter Payload Bay (PLB). It is requested that functional and verification tests on FS-2 be allowed to take place before and after EMI/EMC testing. This will permit the FS-2 program to determine the source of any anomalies.

### 3.4.4 Servicing Operations (GSFC/KSC)

Servicing operations for FS-2 include removal of solar panel covers, solar cell cleaning and battery charging and battery conditioning. Solar cell cleaning will take place after removal of the Plexiglas solar panel covers prior to integration of FS-2 into the PES container.

Maximum FS-2 battery charge characteristics are shown in Table 8. The batteries, once charged, will be acceptable for approximately 45 days. However, battery top charging is desirable whenever the schedule permits, even if the maximum interval between charges has not been reached. Charging/discharging of the payload batteries will be conducted through a Switch Test and Battery Charging port, which is a female DB-9 connector, located at the top of the spacecraft as shown in Figure 18. Equipment for charging the batteries will be provided by the FS-2 team.

**Table 8: FS-2 Battery Charging Characteristics**

| Charge Characteristic | FS-2 |
|---|---|
| Charge current (mA): | 1.0 A |
| Charge voltage (V): | 13 V |
| Charge time (h): | 4.0 h |

Notes for Table 8:

1. 7x SanyoN4000DRL NiCd cells (series)
2. All batteries may be charged through port as shown in Figure 18.
3. Battery conditioning will consist of 2-3 discharge/charge cycles. It is anticipated that this procedure will only be needed once after the planned EMI testing at GSFC.

### 3.4.5 Final Flight Preparation (KSC)

Final flight preparations include ground operations conducted just prior to rollout.

In the OPF, FS-2 battery top charging is requested as late as possible, but at a minimum the battery must be recharged every 45 days. Once the shuttle is at the launch pad, top charging of the FS-2 batteries is requested only if 45 days have passed since the previous charge and if the payload bay is accessible for other reasons. It is requested that access be provided to allow battery top charging both in the OPF and on the PAD should it become necessary. As stated in Section 3.4.4, access to batteries is possible through Switch Test/Battery Charge port on the top of the spacecraft (see Figure 18).

Once proper operation is verified and final battery charging is complete, the three remaining unsealed ports (Switch Test and Battery Charging port, CAN/TTL pot, and Solar Simulator Power port) will be sealed plastic D9 connector covers secured with RTV.

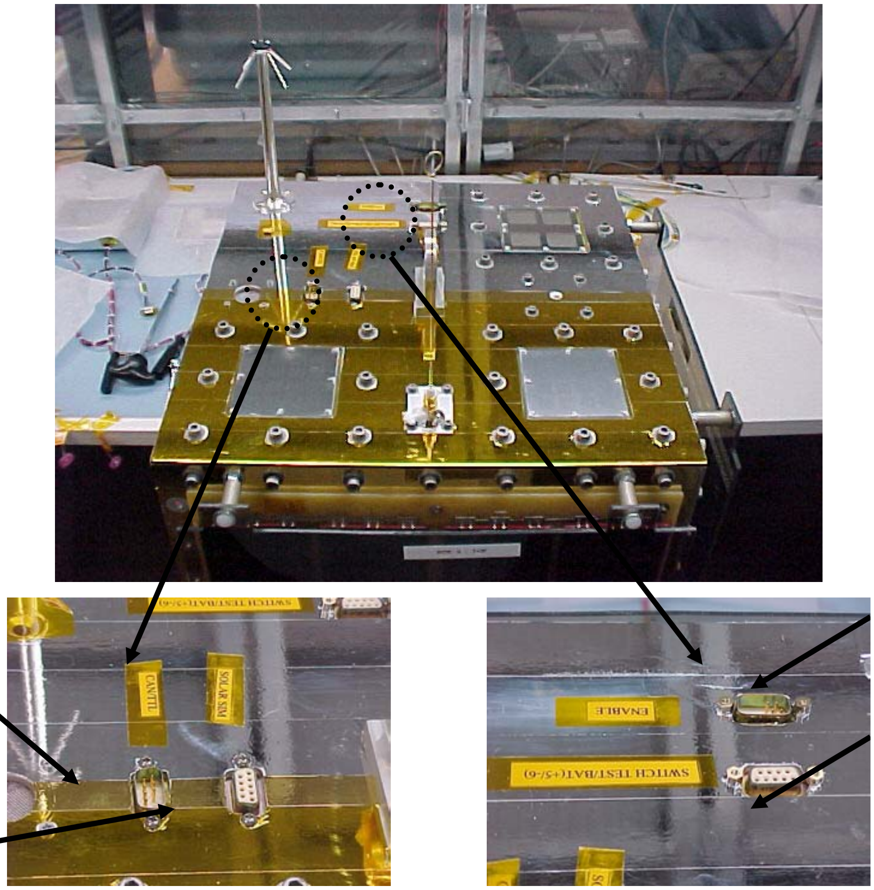


**Figure 18: FS-2 Access for Servicing and Functional/Verification Testing**


**Figure 19. ENABLE Plug**

### *3.4.6 On-PAD Battery Charging*

Battery charging on the PAD will only be required if access to the payload bay is available and greater than 45 days have elapsed since the last battery charging. Battery charging equipment and procedures will be the same as described in Sections 5.3 and 5.4 of the Ground Safety Data Package (GSDP). One additional procedure that will be required is the removal and re-installation of the Switch Test/Battery Charging Port Cover. The procedures for these will be as follows:

1. Attach tether to Switch Test/Battery Charging Port Cover
2. Secure other end of tether to operator
3. Carefully remove RTV securing Switch Test/Battery Charging Port Cover to spacecraft
4. Remove Switch Test/Battery Charging Port Cover
5. Complete battery charging procedures as described in Section 5.4 of the GSDP
6. Re-instert Switch Test/Battery Charging Port Cover
7. Secure Switch Test/Battery Charging Port Cover to spacecraft with RTV
8. Remove tether

# 4. MISSION OPERATIONS REQUIREMENTS

## 4.1 Operational Scenario

### *4.1.1 Deployment Time*

FS-2 should be deployed as soon as possible into the mission, within 5 to 15 minutes after orbit sunrise. This corresponds to time T0.

### *4.1.2 Orbit Parameters*

It is understood that most orbital parameters will be constrained by the primary mission to service the ISS. For maximum orbit lifetime, deployment should be made at the highest possible altitude.

### *4.1.3 Orbiter Pointing*

The Orbiter payload bay should be oriented such that the sun will fully illuminate at least one FS-2 solar panel at the time of deployment. This translates to an Orbiter Z-axis of 90° ±10° with respect to the sun vector. If possible, deployment should be in a generally posigrade direction.

### *4.1.4 FS-2 Ejection from PES*

Once the Orbiter has achieved the required minimum orbit and pointing attitude, ejection of FS-2 may begin. FS-2 should be ejected at a relative speed of ~3 ft/sec. Ejection should take place no sooner than 5 min after orbital sunrise and no later than 15 min after orbit sunrise. Releasing the system at this time will ensure sunlight is on the solar panels producing the required minimum voltage and to allow the system to be observed and photographed during daylight as well as to permit FS-2 to begin battery charging at >T0 (until then, the FS-2 battery is prevented from charging by the use of four microswitch electrical inhibits).

There are no requirements for deployment over CONUS.

## 4.2 Experiment Power

Not Applicable

## 4.3 Experiment Commanding

Not Applicable

## 4.4 Experiment Telemetry

Not Applicable

## 4.5 Crew Involvement

Crew involvement is limited to the following:

- Preparation for and initiation of deployment of FS-2 from HH canister.
- Providing video and still images of FS-2 prior to separation from PES. Views of the forward and aft aspects of the payload are requested. This activity is requested in order to document the physical condition of the payload prior to release from the payload bay.
- Providing video and still images of FS-2 during the separation event. Views of the forward and aft aspects of the payload are requested. This activity is required in order to document the physical condition of the payload upon release from the payload bay as well as to document the operation of the PES separation system.

## 4.6 Instrument Field of View

Not Applicable

## 4.7 Contamination Constraints

FS-2 is inactive while it is stowed in the Orbiter. It is therefore insensitive to payload bay lights and other electrical activity occurring in the orbiter. EMC characteristics will be confirmed prior to launch. Since experiments will not be conducted until the payload is beyond the influence of the Orbiter, vibration effects due to RCS activity is not considered to be an issue. In addition, the risk of liquid or particulate contamination is considered low, since the payload is partially shielded from the environment via the canister. However, during its deployment, FS-2 will be exposed to physical contamination, and use of the RCS should be limited to avoidance maneuvers. Water dumps and flash evaporator operations should also be avoided during deployment.

## 4.8 Air/Ground Communications

Request that Air-to-Ground communications be made available to USAFA during deployment to enhance student involvement with the project.

## 4.9 Customer Supplied Ground Support Equipment

FS-2 CGSE preliminary requirements are summarized in Table 9. Section 5 of the GSDP outlines in detail the specific hardware components that comprise the CGSE. In summary, the major components are three laptop computers, one RF test set, one oscilloscope, one DC power supply, one satellite demodulator, one integrated “Silver Suitcase” which contains the remainder of the RX/TX interface hardware, one Inhibit Switch Verifier, and a programmable battery charger.

**Table 9: Customer Ground Support Equipment Characteristics**

| Item | GSE Characteristic | Applicability |
|---|---|---|
| 1 | Weight | 150 lbs |
| 2 | Number of assemblies | 6-7 |
| 3 | Power required | 115-120VAC |
| 4 | Floor space required | 10' x 20' |
| 5 | CGSE will generate uplink commands | YES |
| 6 | CGSE will receive low-rate customer data | YES |
| 7 | CGSE will receive medium-rate data | YES |
| 8 | CGSE will receive attitude data | NO |
| 9 | Number of standard 15a, 115vac, 60-hz outlets | 2-3 |

Note: All items are Payload specific/unique GSE and do not require GSFC HH GSE

### 4.10 GSFC Payload Operations Control Center (POCC) Requirements

The GSFC POCC will be staffed by FS-2 personnel until deployment. FS-2 personnel will require E-mail/Internet access and long-distance phone access.

### 4.11 Post Mission Data Products

FS-2 requests the following post mission data products:

- Orbit attitude data (at deployment, LVLH, pitch, roll and yaw); state vector (pre-/post-deployment)
- Ejection film and video
- Audio of deployment timeframe: crew cabin and air-to-ground.

## 5. SAFETY ASSESSMENT/ANALYSIS

The general approach to safety assessment has been to examine FS-2 at both the payload and subsystem levels to assess compliance with the safety requirements of NSTS 1700.7B and NSTS/ISS 18798. At the payload level, the analysis identifies top-level requirements that have general applicability to the entire FS-2 spacecraft. In addition to consideration of the specific FS-2 configuration, the payload-level assessment addresses the application of Pre-Approved Hitchhiker Hazard Reports, operational hazards, and interfaces.

At the subsystem level, each FS-2 subsystem is assessed individually for compliance with system-specific safety requirements in order to identify hazards that are unique to each subsystem. At both the payload and subsystem levels, potential hazards have been identified and controls have been established based on the safety requirements or guidance documents. The safety assessment has resulted in implementation of fault tolerant and minimum risk designs to control hazards at both the payload level and the subsystem level. The hazard reports applicable for the FS-2 mission, including several payload-unique hazard reports, have been generated and are attached in Appendix A. Unique hazard reports are called out throughout the assessment in the sections for which they apply.

## 5.1 Payload-Level Safety Assessment/Analysis

This section will address those safety concerns that apply to the FS-2 payload as a whole.

### *5.1.1 Interfaces*

FS-2 requires no services from the Orbiter, therefore, the only interface between FS-2 and the Orbiter is the mechanical interface with the PES. FS-2 will be safe without services in accordance with NSTS 1700.7B, paragraph 200.4a.

### *5.1.2 Operations*

FS-2 is designed to be completely inactive while on the Orbiter. Section 218 of NSTS 1700.7B was reviewed and does not apply to FS-2. Commanding of FS-2 is not possible until inhibits in the FS-2 electrical systems are removed upon separation from the PES. Prior to deployment of FS-2 from the PES, hazardous systems, such as the battery, will maintain fault tolerance. All safety features are designed for launch, landing, on-orbit, and emergency landing phases of Shuttle flight.

In accordance with NSTS 1700.7B paragraphs 215.1 and 215.2, the post-ejection operational scenario has been examined to identify planned flight operations that pose re-contact hazards. Following FS-2 ejection from the Orbiter via the PES, the satellite will move away from the Orbiter, and since FS-2 has no propulsion capabilities, no unique re-contact hazard has been identified.

### *5.1.3 Structure/Fracture Control*

The major structural components on FS-2 are described in Section 2.3.1. As described, all major structural components are made from stress-corrosion resistant 6061-T6 and 6061-T651 aluminum. NSTS 1700.7B contains the following paragraphs that specifically apply to structures and fracture control:

- 208.1 Structural Design
- 208.2 Emergency Landing Loads
- 208.3 Stress Corrosion

FS-2 submitted Pre-approved Hazard Report HH-F-5, *Failure of Hitchhiker Payload Structure*, to demonstrate compliance with 208.1, 208.2, and 208.3. FS-2 has no pressure vessels; therefore, paragraph 208.4 of NSTS 1700.7B does not apply. FS-2 has no sealed compartments; therefore paragraph 208.5 does not apply. Where necessary, analysis was performed for compartments formed by structure, electronics boxes, and mechanical housings to ensure that adequate venting exists. Details of the FS-2 approach to structural design are provided in the Structural Verification Plan (SVP) and Structural Verification Report (SVR). FS-2 was designed for fracture control as required by paragraph 208.1. A Fracture Control Plan (FCP) and Fracture Control Report (FCR) are being submitted to fulfill the requirements of NASA-STD-5003.

Details of fastener integrity and backout protection compliance, as well as verification results, and designs are provided in the SVR.

### *5.1.4 Materials*

NSTS 1700.7b contains several materials safety requirements that may be addressed for FS-2 as a whole. These paragraphs are:

- 209.1 Hazardous Materials

- 209.1a Fluid Systems
- 209.1b Chemical Releases
- 209.2 Flammable Materials
- 209.2a Orbiter Cabin
- 209.2b Other Habitable Areas
- 209.2c Outside Habitable Areas
- 209.3 Materials Offgassing in Habitable Areas

FS-2 contains no fluids or chemicals, nor will it release or eject any materials in or near the Orbiter, therefore, paragraphs 209.1a and 209.1b do not apply. FS-2 will demonstrate compliance with paragraph 209.2 by submitting Pre-approved Hazard Report HH-F-4, "Flammable Materials". Note that the only sub-paragraph of 209.2 that applies is 209.2c "Outside Habitable Areas" because no component of FS-2 is installed in the Orbiter Cabin or Habitable Areas. For the same reason, paragraph 209.3 does not apply. All materials used in the FS-2 payload will have an "A" rating for flammability as defined in MSFC-HDBK-527F/JSC 09604.

### *5.1.5 Electrical*

FS-2 contains electrical systems for distribution of power and signals to subsystems and experiments as described in Section 2.3. The power sources associated with these systems are solar panels and a battery. The battery will be addressed later in the document at the subsystem level.

It is important to reiterate general features of FS-2 that eliminate certain common electrical system hazards. First, there is no electrical interface between FS-2 and the Orbiter, therefore, improper design of FS-2 circuitry cannot have a hazardous effect on Orbiter electrical systems.

Two operations may require use of the FS-2 electrical systems on the ground while in the payload bay: battery charging, and final status checks of electrical system safety inhibits. These operations will be conducted as close to launch as possible. The electrical system will be inhibited to prevent activation of hazardous functions while in the Orbiter. The electrical power system will employ a system of fault tolerant electrical inhibits that prevent power from reaching any subsystems during launch, landing, and on-orbit. These inhibits will not be removed until the payload has been deployed from the Orbiter. The electrical inhibits schemes used on FS-2 is compliant with NSTS 1700.7B, Section 201 and is discussed in Section 2.3.2. The purpose of mentioning the inhibits in this section is two-fold 1) to indicate that all electrical power systems will be disabled until after deployment and 2) to consider their presence while assessing the possible effects of electrical system design flaws.

Having noted the above design and operational features, the two electrical system hazards are loss of electrical system safety inhibits or ignition of a flammable payload bay (PLB) atmosphere due to improper electrical system design. Electrical system hazards to personnel who may be performing charging or inhibit status checks prior to launch such as shock, hot surfaces, or toxicity will be addressed as a ground safety hazard.

Electrical failure due to improper circuit protection or wire sizing is a potential cause for loss of electrical system inhibits. In order to address the hazard of removal of safety inhibits due to electrical system faults, FS-2 has submitted Pre-approved Hazard Report HH-F-1 *Damages to STS Electrical Systems*. All FS-2 electrical systems are in compliance with NSTS 1700.7B, section 213 *Electrical Systems*, and NST 18798, TA-92-038 *Protection of Payload Electrical Circuits*. Paragraph 213.1, "General" of NSTS

1700.7B, and TA-92-038, provide requirements for proper wiring design and circuit protection. GSFC Code 870 will perform or concur with payload electrical power distribution design review/assessment and perform inspection to verify compliance with TA-92-038 letter.

Another potential cause of loss of electrical system inhibits is susceptibility to EMI. NSTS 1700.7B, Paragraph 213.3, "Lightning", addresses the possibility of electrical circuit susceptibility to electromagnetic effects caused by lightning. Note that, in accordance with 213.3, the electrical system safety inhibits will be mechanically activated separation micro-switches that cannot be removed by electrical surges, such as those due to lightning strikes. To address paragraph 213.1 and EMI susceptibility in general, FS-2 will be subjected to EMI susceptibility testing by GSFC once integrated into the Hitchhiker/PES system. GSFC will submit those results separately as part of the integrated safety package.

The micro-switches used are Honeywell 1HE1-6. They are hermetically-sealed limit switches with bottom receptacle terminal plunger actuation. The HE Series Hermetically-Sealed Limit Switch meets hermetic sealing as defined in MIL-S-8805, Symbol 5. The HE Series features true hermetic sealing (metal-to-metal, glass-to metal construction), ensuring maximum sealing effectiveness for exceptionally long periods of time regardless of constant changes in atmospheric pressures and temperatures. Table 10 lists product specifications for these micro-switches.

**Table 10: Micro-switch Product Specifications**

| | |
|---|---|
| Operating Force (O.F.) | 26.69 N-53.38 N [6 lbs-12 lbs] |
| Full Overtravel Force | 133.45 N max. [30 lbs max.] |
| Release Force (R.F.) | 17.79 N min. [4 lbs min.] |
| Pretravel (P.T.) | 1.016 mm max. [.040 in max.] |
| Overtravel (O.T.) | 6.35 mm min. [.250 in min.] |
| Differential Travel (D.T.) | 0.508 mm max. [.020 in max.] |
| Shock | 200 g, .007 sec, half sine pulse |
| Vibration | 10 Hz – 2000 Hz – 500 Hz at 20 gs |
| Operating Temperature Range | -55 deg C to 125 deg C [-67 deg F to 257 deg F] |

To address the hazard of ignition of a flammable PLB atmosphere, FS-2 has submitted Pre-approved hazard report HH-F-3 *Ignition of a Flammable PLB Atmosphere*. No surface on FS-2 will exceed 352° F as required by Interpretation Letter NSTS 18798, NS2/81-M082.

### 5.1.6 Mechanical

FS-2 will have no mechanisms of any kind. Therefore, no *Mechanical Systems Verification Plan* (MSVP) was submitted.

### 5.1.7 Thermal

The FS-2 thermal model, written in MatLab/Simulink, is fully described in Appendix G. It was test-verified during Thermal/Vacuum testing of both the FS-2 Engineering Model and the FS-2 Qualification Model. Results from FS-2 Qualification Model thermal/vacuum testing are also described in Appendix G.

From a Shuttle safety standpoint, the most important output of this model is the coupled analysis for behavior of key FS-2 components during worst-case conditions prior to deployment. This analysis will formally be done using the SINDA thermal modeling code by GSFC thermal engineers. As a preparation for this, a preliminary model of the case of FalconSat-2 coupled to the STS through PES was created.

The inputs to this thermal model are conduction value and boundary temperature values obtained from the thermal engineers at GSFC.

The boundary temperatures are from three attitude situations for the shuttle:

- Bay-to-sun case (worst-case hot)
- Bay-to-Earth case
- Tail-to-sun case (worst-case cold)

To bound the problem, the worst-case hot and cold cases were analyzed. For this, the conductors and boundary temperatures for the bay-to-space case for the worst-case-cold situation were implemented. This is because the shuttle could theoretically have its bay pointed toward space for an indefinite period of time. For the worst-case-hot case, the conductors and boundary temperatures for the bay-to-sun case for 60 minutes and the bay-to-earth case for 30 minutes per orbit were implemented. This is because the shuttle's orbit has a 30 minute eclipse, so the bay-to-sun case can only happen for a maximum of 60 minutes per orbit, and during the other 30 minutes of the orbit, the worst-case-hot will be the bay-to-earth case. The model was run and the point at which the batteries reached their storage temperature limits was determined. The results are summarized below:

- Bay-to-sun case (worst-case-hot):
- The batteries reach the maximum storage limit of 50°C in 240 minutes (in the third orbit cycle of a worst-case-hot condition)
- The module boxes reach the maximum storage limit of 50°C in 150 minutes (in the second orbit cycle of a worst-case-hot condition)

Results are shown in Figure 20.

- Tail-to-sun case (worst-case-cold):
- The batteries never reach the minimum storage limit
- The module boxes never reach the minimum storage limit

Results are shown in Figure 21.


Use or disclosure of data contained on this sheet is subject to the restrictions on the title page of this document.

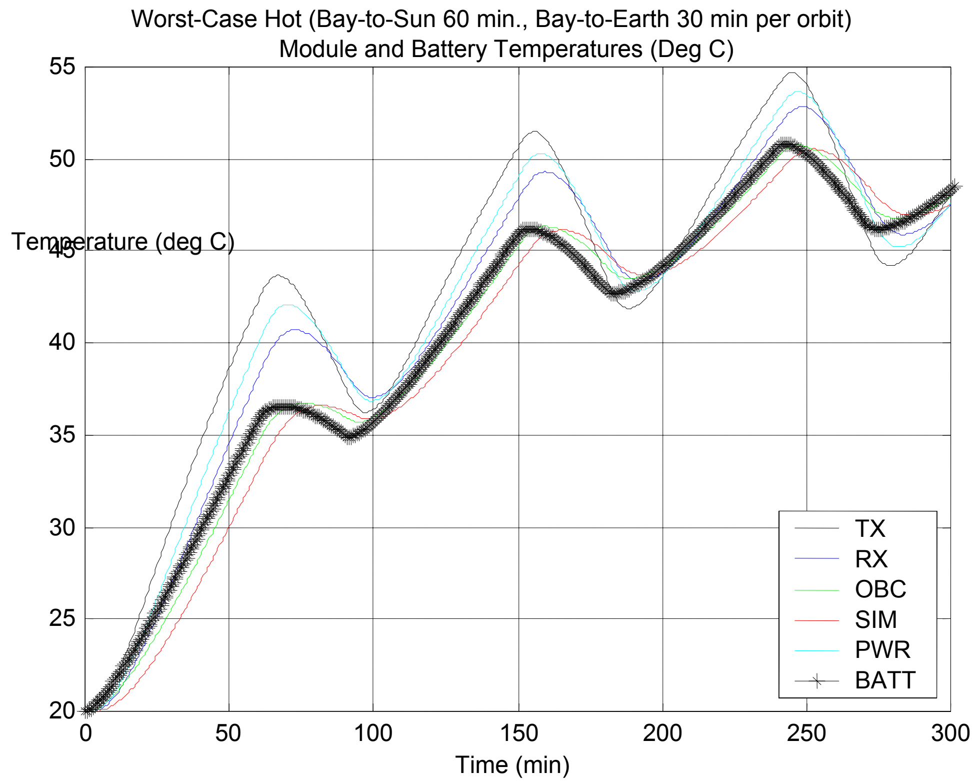


**Figure 20: Bay -to-sun coupled case (Worst-case hot coupled thermal behavior)**

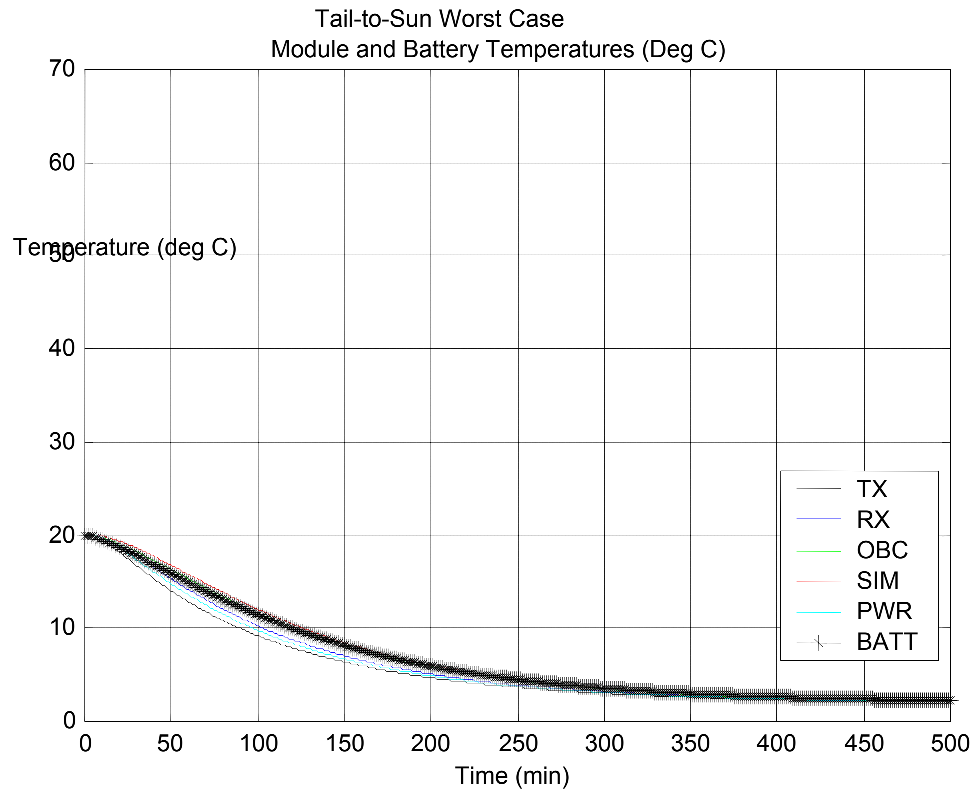


**Figure 21: Tail-to-sun coupled case (Worst-case cold coupled thermal behavior)**

### *5.1.8 Electromagnetic Interference*

Pre-Approved Hazard Report HH-F-2 *Electromagnetic Interference with Space Shuttle Operations* was submitted for FS-2. Payload electrical/electronic system was designed not to radiate above, conduct above, or be susceptible to the radiated or conducted EMI levels specified in ICD-2-19001 limits. GSFC Code 870 will perform EMI tests of the fully integrated HH payload to ICD 2-19001 limits. Any payload specific EMI exceedences to the ICD-2-19001 levels will be documented in the payload-unique ICD, and a JSC approval will be obtained.

### *5.1.9 Grounding/Bonding*

The FS-2 electrical power subsystem, as described in Section 2.3.2, is grounded to the FS-2 structure. The entire spacecraft structure, including all electronics trays, provides for continuous bonding through metal-to-metal contact. The 6061-T6 Al structure was alodined to improve electrical conductivity. Figure 22 is a diagram of FS-2 bonding and grounding.

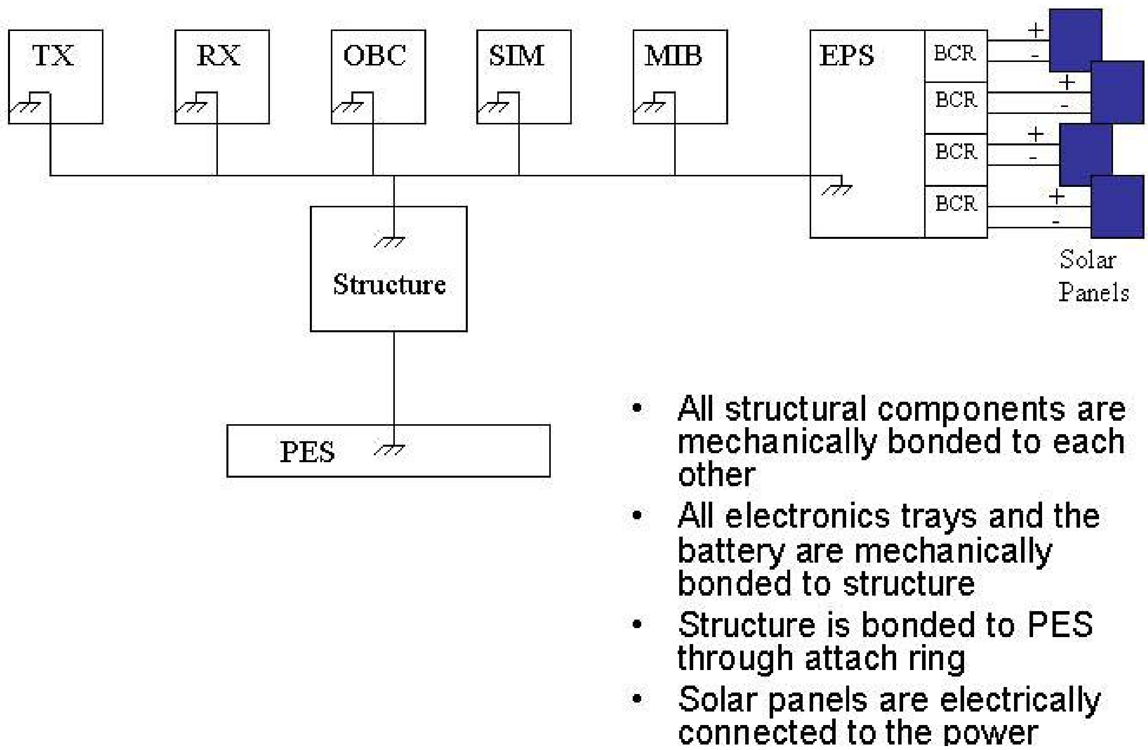


**Figure 22: Bonding and Grounding**

As specified in Section 5.1.3, HH-F-3 *Ignition of PLB Atmosphere* was submitted for FS-2. The hazard controls specified therein cover grounding of FS-2 to Hitchhiker grounding, and Hitchhiker to Orbiter grounding. FS-2 meets the requirements of ICD 2-19001 specified as hazard controls in HH-F-3. Section 11 of the Battery Acceptance Report located in Appendix E contains further bonding and grounding information.

### *5.1.10 EVA Hazard*

The FS-2 payload was analyzed and designed to preclude inadequate structural design resulting in sharp edges and protrusions in order to protect ground personnel and, in the event of the Hitchhiker motorized door assembly (HMDA) failure, those personnel conducting unplanned EVAs. Hazards caused by inadequate structural design was precluded by meeting sharp edge requirements of NSTS 07700, Vol. XIV, Appendix 7 for all accessible areas of the payload (*i.e.*, the top surface of the FS-2 satellite).

### *5.1.11 Pre-approved HH Hazard Reports*

FS-2 has submitted the following pre-approved HH Hazard Reports:

- HH-F-1 *Damage to STS Electrical Systems*
- HH-F-2 *Electromagnetic Interference with Space Shuttle Operations*
- HH-F-3 *Ignition of a Flammable Payload Bay Atmosphere*
- HH-F-4 *Flammable Materials*
- HH-F-5 *Failure of Hitchhiker Payload Structure*

FS-2 has submitted the following unique hazard reports:

- FS2-F-1 *FS-2 Battery Leakage/Rupture*
- FS2-F-2 *FS-2 Radio Frequency Interference with Space Shuttle Operations*

### *5.1.12 Verification of Hazard Controls*

Subsystem level tests and analysis were performed by USAFA and subsystem manufacturers, as required, to verify hazard controls. Details of verification tests and analysis are given in the hazard reports in Appendix A. All verifications performed at the subsystem level were conducted by USAFA, with review and concurrence by GSFC, to ensure that the design, as built and tested, complies with safety requirements. The payload complies with the requirements of NSTS 18798, TA-94-018, *Safety Policy for Detecting Payload Design Errors*.

## 5.2 Subsystem-Level Safety Assessment/Analysis

### *5.2.1 Electrical Power Subsystem*

The FS-2 electrical power subsystem consists of solar panels, battery charge regulators, power conditioning and distribution, associated wiring and connectors, and a single battery, as described in Section 2.3.2. The general issues of electrical system design safety as it applies to FS-2 were covered in Section 5.1.5. Therefore, there will be no further discussion of general design issues in this section. This section will focus on the primary safety-related issue of the associated with the electrical power subsystem, the battery.

The requirements of NSTS 1700.7B paragraph 213.2 were reviewed to assess the hazards associated with the FS-2 battery and their potential causes. Guideline hazard report GHR-10 of JSC 26493 was also reviewed for the same purpose. Typical battery hazards and their causes were identified and analyzed for applicability to the FS-2 battery. The hazard associated with the FS-2 battery can be described as release of explosive gasses and/or electrolytes that can lead to fire, explosion, corrosion, contamination, potential injury to the crew, damage to the Orbiter, or damage to other payloads. The FS-2 battery will not be operated until deployed from the PLB due to the inhibits in the FS-2 power supply system described in Section 2.3.2. The first time that the FS-2 battery will operate is upon deployment from PES.

Design features of the battery box and cells will control Battery hazards, and their causes. JSC 20793 was consulted for guidelines as to appropriate controls for the various hazard causes that were identified. Battery safety issues are addressed in hazard report FS2-F-1 *FS-2 Battery Leakage/Rupture.* This hazard report is located in Appendix C. These hazards, their causes and controls are summarized in Table 11.

Information on battery verification plan as well as tests are included in HR and HV.

### 5.2.2 Structure Hazards

Hazards posed by the FS-2 structure are addressed at the payload-level in Section 5.1.3.

### 5.2.3 Attitude Determination and Control Subsystem Hazards

The FS-2 ADCS is described in Section 2.3.6. There are no potential hazards posed by this system.

### 5.2.4 Data Handling Subsystem Hazards

The FS-2 Data Handling Subsystem is described in Section 2.3.3. A review of the C&DH subsystem and the safety requirements of NSTS 1700.7B indicates that there are no potential hazards associated with this system. Because of the EPS inhibits scheme, described in Section 2.3.2, the Data Handling subsystem will be un-powered until separation from the PES. Therefore, no specific hazards for this subsystem will be addressed.

**Table 11: FS-2 Battery Hazards and Controls**

| Hazard Cause | Control |
|---|---|
| Shorting, Internal to cells | Acceptance testing of cells for electrical defects. |
| Shorting, Internal to battery box. | Internal surfaces of the battery box are coated with electrolyte resistive, non-conductive coating.<br>Wiring insulated, stress-relieved and staked to prevent motion. |
| Shorting, External | 3-A Polyswitch installed in FS-2 power supply system in accordance with the requirements for circuit protection as referenced in TA-92-038. |
| Cell reversal/Over-discharge | Battery cells matched to prevent uneven discharge. |
| Excessive Internal Cell Pressure | Cells are vented.<br>Cells are aligned such that vents are on Y axis.<br>Cell vents tested by manufacturer. |
| Over-temperature | Environmental: FS-2 is designed with coatings to maintain battery temperature within optimal temperature limits of –20 °C to 30 °C during all phases of flight, as determined by analysis and test. Thermal analysis and test were conducted by USAFA and GSFC. Thermal tests were conducted by USAFA.<br>Shorting: See Shorting entries in this Table.<br>Charge/Discharge: There will be no on-orbit charging or discharging of the battery until deployed from the PLB. |
| Freeze/Thaw | The FS-2 Battery box is designed with insulation, coatings, and paints required to maintain battery temperature within the temperature limits of –20° C to 30° C during all phases of flight, as determined by analysis and test. Thermal analysis and test are being conducted by USAFA and GSFC. |
| Accumulation and ignition of Hazardous gas mixture | Battery box will have four filtered vents to prevent accumulation of gas. |
| Leakage of electrolyte | Battery box vents will have filters. Void spaces of the battery box are filled with Pigmat absorbent material. Battery box interior has an anti-corrosive coating. The use of a Sigaflex graphite gasket between the case and any lids shall provide a leak tight seal, but maintain electrical continuity. |

### *5.2.5 Communication Subsystem Hazards*

For safety analysis, the communications system must be considered on the basis of its components, which are electrical, mechanical, and structural, and its functions while operating. There is no means for moving, deploying, or retracting the S-band and VHF antennas. These components are designed to be Low-Risk fracture parts.

When operating, the communications system is capable of producing RF radiation, and therefore must comply with paragraph 202.5 of NSTS 1700.7B, which references ICD-2-19001 for RF-level safety requirements. This paragraph identifies RF transmission as a catastrophic hazard unless proven otherwise. Chapter 10.7.3 (Cargo-Produced Interference Environment) of the ICD contains the applicable requirements. The only unique hazard associated with the FS-2 communications system is RF transmission. Paragraphs 10.7.3.2.2.1 and 10.7.3.2.2.2 of the ICD provide maximum limits for intentional and unintentional radiated electric fields. For the purpose of safety analysis, FS-2 is assumed to be an intentional RF radiator. In the case of intentional radiated fields, the ICD provides limits for payload-to-payload, payload-to-Orbiter, and payload-to-RMS. Paragraphs 202.5a, 202.5b, and 202.5c provide requirements for inhibits based on levels of compliance with the ICD requirements.

The following analysis and calculation demonstrates compliance of the communications subsystem with RF safety requirements. All analysis assumes worst-case, i.e. maximum RF transmission cases. Actual operational RF radiation will be less than or equal to this. Specifically, the following assumptions were made:

- Maximum transmitter output RF power = 0.5W
- Maximum antenna gain = 3dB
- Line losses between transmitter and antenna = 0dB

The electric field intensity is computed by using the following equation:

$$E = \sqrt{120\pi P} = \frac{\sqrt{30P_t}}{R}$$

where: $P$ = Power density (W/m$^2$)

$E$ = Electric Field Intensity (V/m)

$R$ = Distance from radiation source (m)

$P_t$ = Transmitted Power (W)

Assuming a best antennae gain of 3dB the, effective transmitted power would be 1.0 W, representative of the FS-2 S-band directional power, $E$ at a distance of 0.25 m from the source would be:

$$E = \sqrt{30(1.0)}\big/0.25 = 21.91 \text{ V/m}$$

Converting to dB:

$$E = 20\log\left(\frac{21.91}{10^{-6}}\right)\frac{\text{dB}_{\mu\text{V}}}{\text{m}} = 146.8\,\frac{\text{dB}_{\mu\text{V}}}{\text{m}}$$

The electric field intensity as a function of distance from source is illustrated in Figure 23.

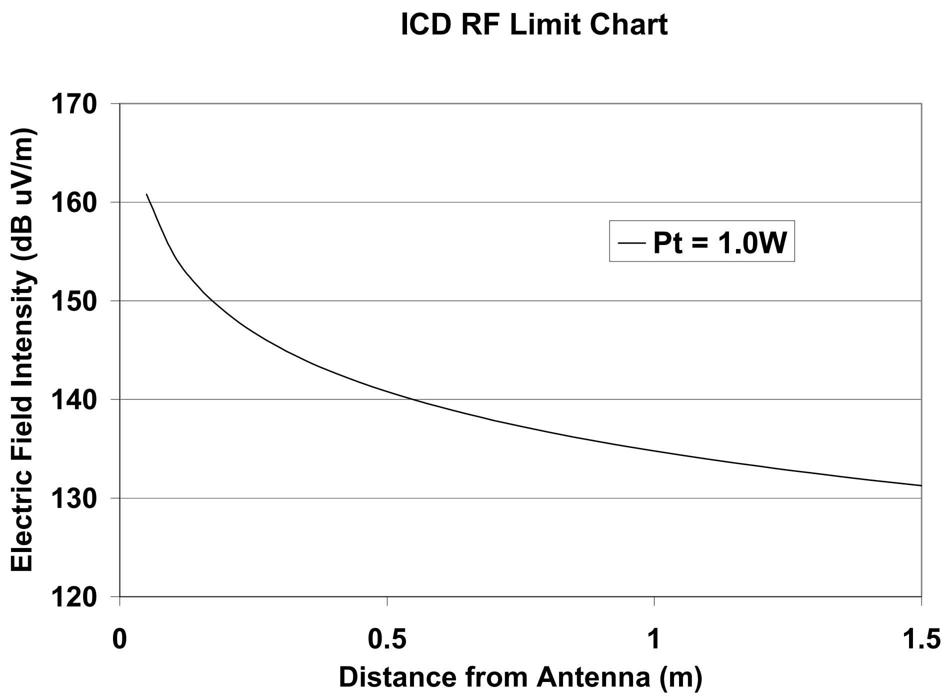


**Figure 23. ICD RF Limit Chart.**

Electric field intensity is plotted as a function of distance from a radiating source with effective transmitting power of $P_t$ = 1.0 W.

To verify the assumptions made in the above analysis, the FS-2 transmitter output power was measured in the laboratory using a spectrum analyzer. The test set up is shown in Figure 24. A photograph of the measured signal is shown in Figure 25. The known line losses in the test set-up, including a 20dB attenuator and a T-connector, provide a total signal attenuation of -27 dB. Figure 25 shows that the peak of signal is at -30dB. Therefore, the transmitter output power is -3dB which corresponds to 0.5W. Due to complexity and cost, the actual antenna gain was not measured. However, the antenna uses a classic ground plane design used throughout the industry with a known maximum gain of 3 dB in the peak lobe.

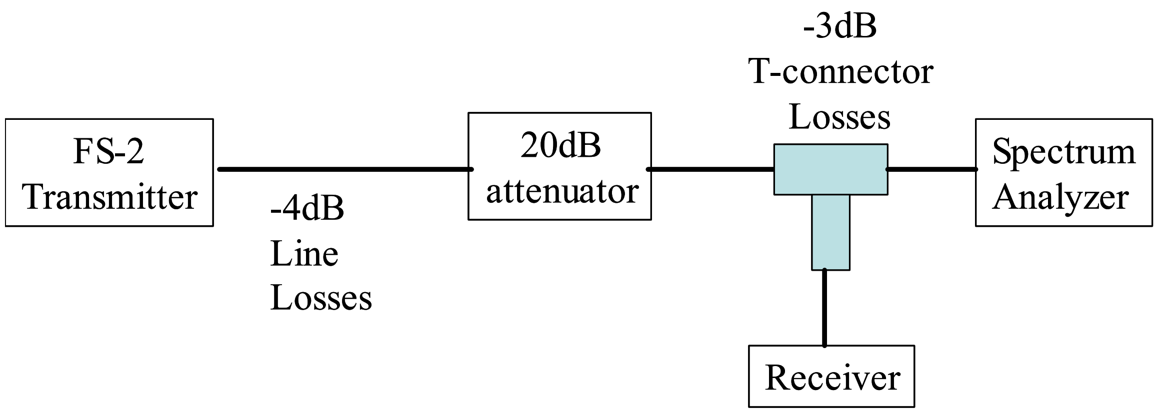


**Figure 24: FS-2 Transmitter Output Test Set-Up**

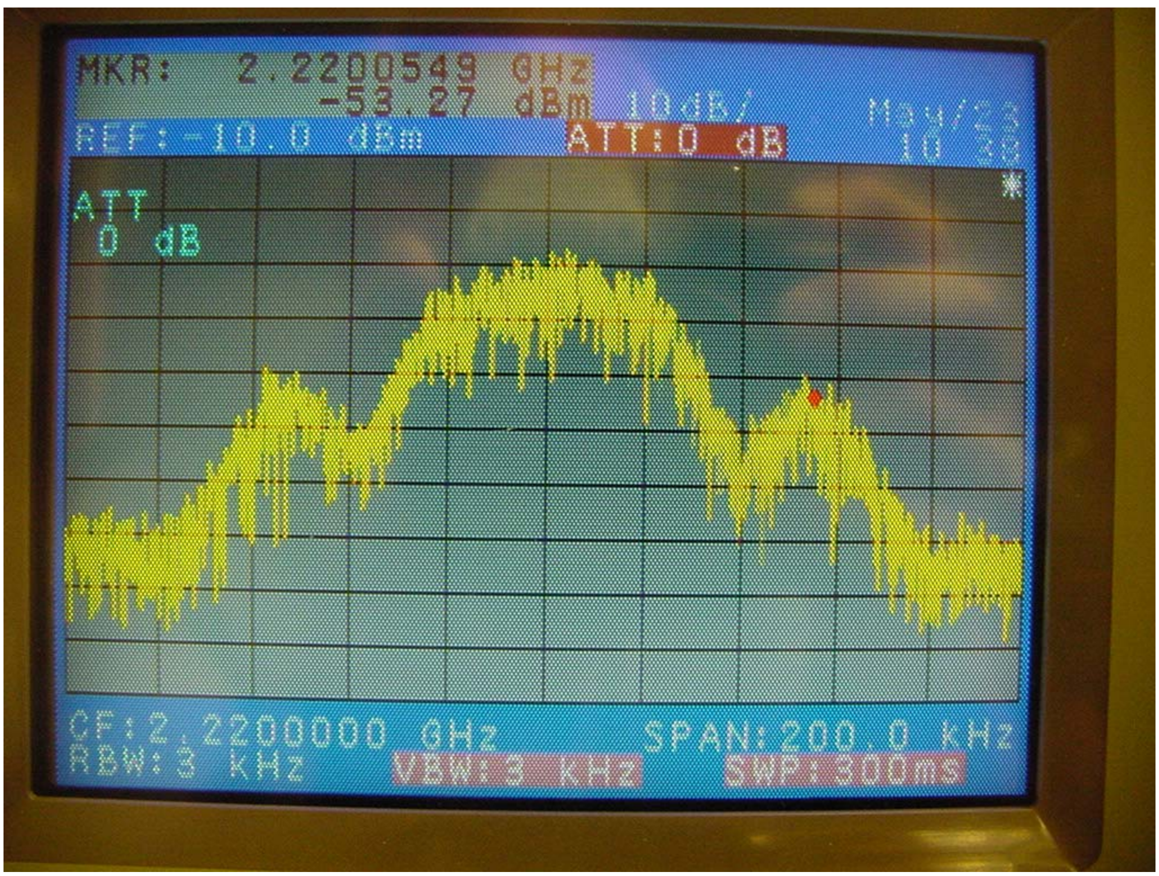


**Figure 25: Photograph of measured FS-2 transmitter output spectrum.**

A summary of requirements and plans for conformance appears in Table 12. Hazard Report FS2-F-2, *FS-2 Radio Frequency Interference with Space Shuttle Operations* has been developed to address the issue of inadvertent RF transmission.

**Table 12. ICD RF Limits: Requirements and Conformance Plans.**

**REQUIREMENTS:**

ICD 2-19001 Chapter 10.7.3 (Cargo-Produced Interference Environment) specifies that RF emissions must be less than 35 $dB_{\mu V}$/m(downlink), 79 $dB_{\mu V}$/m (crosslink), and 76 $dB_{\mu V}$/m (beacon) for unintentional radiation. For intentional radiation, payload-to-Orbiter requirements provide the lowest limit at 150 $dB_{\mu V}$/m for the FS-2 transmission frequencies.

**CONFORMANCE:**

- Transmitter output power was measured to verify RF radiation analysis assumptions.
- The maximum output of the FS-2 transmitters, 147 $dB_{\mu V}$/m at $R$ = 0.25 m and 135 $dB_{\mu V}$/m at $R$ = 1.0 m, as calculated using the equations above, exceeds the ICD limit for unintentional radiation, but it does not exceed the limit for intentional payload to Orbiter radiation (150 $dB_{\mu V}$/m at 2.22 GHz). Thus the communications system RF transmission is a catastrophic hazard as defined by NSTS 1700.7B, section 202. Per 202.5a four independent inhibits must be provided because the ICD limits for unintentional radiation are exceeded.
- The communications system's catastrophic hazard potential is controlled by the FS-2 EPS inhibit scheme. Four inhibits prevent any battery power from being applied to the communications system. A single seperation switch prevents power directly from the FS-2 solar panels to power the transmitter. However, RF transmission levels are 15 $dB_{\mu V}$/m below ICD-2-19001 limits (at $R$ = 1.0 m) with the payload bay doors open and sunlight would only be available to the panels with the doors open and HH lid open.

The RF system does not start transmitting in the HH even with sunlight on solar panels. The inhibits in the EPS will prevent the communications system from receiving power until after FS-2 has been ejected from the Orbiter via the PES.

### *5.2.6 MESA*

The MESA experiment on FS-2 is described in Section 2.3.7. A review of the MESA experiment and the safety requirements of NSTS 1700.7B indicates that there are no potential hazards associated with this system. Because of the EPS inhibits scheme, described in Section 2.3.2, the MESA will be un-powered until separation from the PES. The MESA instruments consist of PCBs with thin stainless steel plates fastened on top. Therefore, no specific hazards for this subsystem will be addressed.